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NAS9-14319

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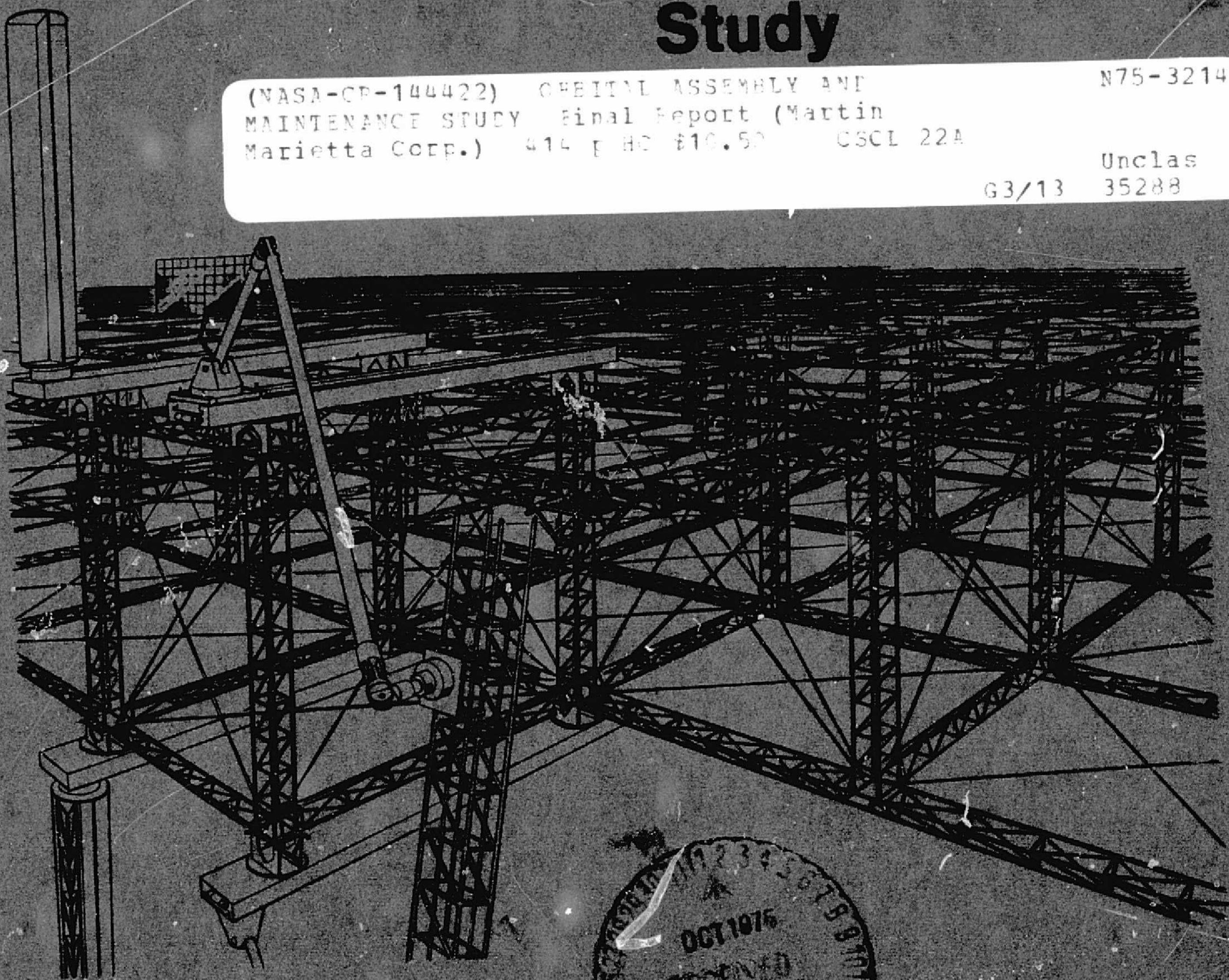
Final Report

August 1975

Orbital Assembly and Maintenance Study



MARTIN MARIETTA

MCR-75-319
Final Report

August 1975

T-1060
Line Item 4

ORBITAL ASSEMBLY
AND
MAINTENANCE STUDY

NAS9-14319

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FOREWORD

This document presents the results of work performed by the Martin Marietta Corporation while under contract to NASA L. B. Johnson Space Center. This final report was prepared as partial fulfillment of contract NAS9-14319, Orbital Assembly and Maintenance Study. The NASA Contracting Officer's Representative was Herbert G. Patterson of the Future Programs Office, Engineering and Development Directorate.

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ACRONYMS AND ABBREVIATIONS

ABS	Absolute
ACS	Attitude Control System
AESPA	Airborne Electronically Steerable S-Band Phased Array
ANT.	Antenna
AOP	Advanced Onboard Computer
AOT	Average Operational Time
APS	Auxiliary Propulsion System
ASSY	Assembly
B	Billion
BOL	Beginning of Life
CFA	Cross Field Amplifier
CG	Center of Gravity
cm	Centimeters
D	Diameter
DC	Direct Current
DEG	Degrees
DEL	Delivery
DIA, DIAM	Diameter
DOD	Department of Defense
DOF	Degrees of Freedom
DWS	Disaster Warning Satellite
EA	Each
ECLSS	Environmental Control and Life Support System
EOGP	Earth Observations Geosynchronous Platform
EOTS	Earth Orbital Teleoperator System
EPS	Electrical Power System
EQUIP	Equipment
EVA	Extravehicular Activity
EXT	External
FREQ	Frequency
FRUSA	Flexible Rolled-Up Solar Array
FT	Feet
FT^2	Square Feet
FT^3	Cubic Feet
F/S	Feet per Second

ACRONYMS AND ABBREVIATIONS (Cont'd)

GAC	Grumman Aerospace Corporation
GH_z	Giga Hertz
GNS	Guidance, Navigation, and Stabilization
GN_2	Gaseous Nitrogen
GS	Ground System
GW	Gigawatts
HDR	High Data Rate
HEO	High Earth Orbit
H_g	Mercury
HPBW	Half Power Band Width
I	Moment of Inertia
IEO	Intermediate Earth Orbit
IN.	Inch
IOSS	Integrated Orbital Servicing Study
IR	Infrared
I_{sp}	Specific Impulse
IUS	Interim Upper Stage
IVA	Intravehicular Activity
JSC	Johnson Space Center
K	Thousand
KG	Kilogram
KM	Kilometer
LBS	Pounds
LCE	Low Cost Expendable
LDR	Low Data Rate
LEO	Low Earth Orbit
LOFT	Low Frequency Telescope
M	Meters, Million, Moment
MA	Mobile Assembler
MAX	Maximum
MCC	Mission Control Center
MDAC	McDonnell Douglas Astronautics Company
MDR	Medium Data Rate

ACRONYMS AND ABBREVIATIONS (Cont'd)

MH_z	Mega Hertz
MIN	Minimum, Minutes
MM	Millimeters
MMC	Martin Marietta Corporation
MMD	Mean Mission Duration
MMU	Manned Maneuveruing Unit
MPS	Main Propulsion System
MPTS	Microwave Power Transmission System
M/S	Meters per Second
MSFC	Marshall Space Flight Center
MSM	Manned Servicing Module
MW	Microwave
NA	Not Applicable
NASA	National Aeronautics and Space Administration
NASCOM	NASA Communications Network
N. MI.	Nautical Miles
NO.	Number
NOAA	National Oceanic Atmosphere Agency
NOCC	Network Operations Control Center
N_2H_4	Hydrazine
OAM	Orbital Assembly and Maintenance
OMS	Orbital Maneuvering System
OS	Operator's Station
PC	Power Conditioner
PF	Parts Factor
P/L	Payload
PRN	Pseudo Random Noise
PROC	Processor
PROP	Propulsion
PSI	Pounds per Square Inch
PUT	Payload Utilization of Tug
PWR	Power
QTY	Quantity

ACRONYMS AND ABBREVIATIONS (Cont'd)

RAE	Range-Azimuth-Elevation
RAT	Radio Astronomy Telescope
RCS	Reaction Control System
RCVR	Receiver
R&D	Research and Development
RDTE	Research, Development, Test and Engineering
REV	Revolutions
RF	Radio Frequency
RI	Rockwell International
RMS	Remote Manipulator System
SA	Solar Array
SAA	Solar Array Assembly
SATS	Satellites
SC	Spacecraft
SEC	Seconds
SEOS	Synchronous Earth Observations Satellite
SEPS	Solar Electric Propulsion Stage
SERV	Servicing
SGLS	Space-Ground Link Subsystem
SMA	Slave Manipulator Arm
SOC	Shuttle Operations Center
SSPD	Space Shuttle Payloads Description
SSPS	Satellite Solar Power Station
STDN	Spaceflight Tracking and Data Network
STS	Space Transportation System
T	Time, Temperature
TBD	To Be Determined
TCCC	Test Condctor's Control Console
TCS	Thermal Control System
TDRS	Tracking and Data Relay Satellite
TDRSS	Tracking and Data Relay Satellite System
TLM	Telemetry
TMTR	Transmitter

ACRONYMS AND ABBREVIATIONS (Cont'd)

TOC	Tug Operations Center
TT&C	Tracking, Telemetry, and Communications
TV	Television
TX	Transmitter
ULT	Ultimate
UOPD	Unmanned Orbital Platform Definition
V	Velocity
VDC	Volts DC
VHF	Very High Frequency
W	Watts, Weight
W_p	Propellant Weight
XMTR	Transmitter
^oF	Degrees Fahrenheit
^oC	Degrees Centigrade
^oK	Degrees Kelvin
Δ	Delta
ΔV	Velocity Change
μ	Micron
μ_p	Mass Ratio
ϕ	Phase Angle
ω	Rotational Velocity
λ	Failure Rate, Wavelength
σ	Stephen-Boltzmann Constant
α	Absorptivity
ϵ	Emissivity

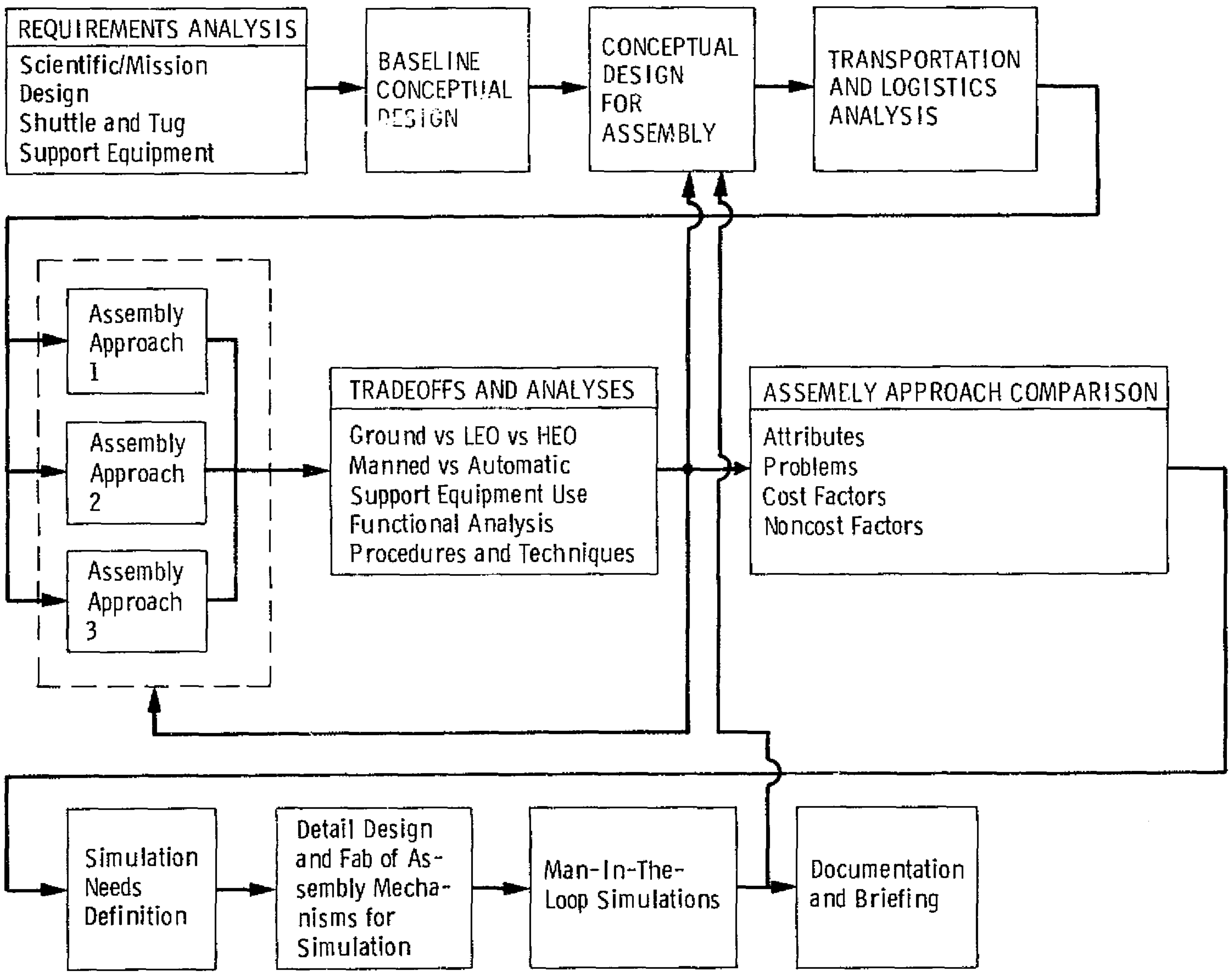


Figure I-1 Study Task Flow - Assembly of Space Systems

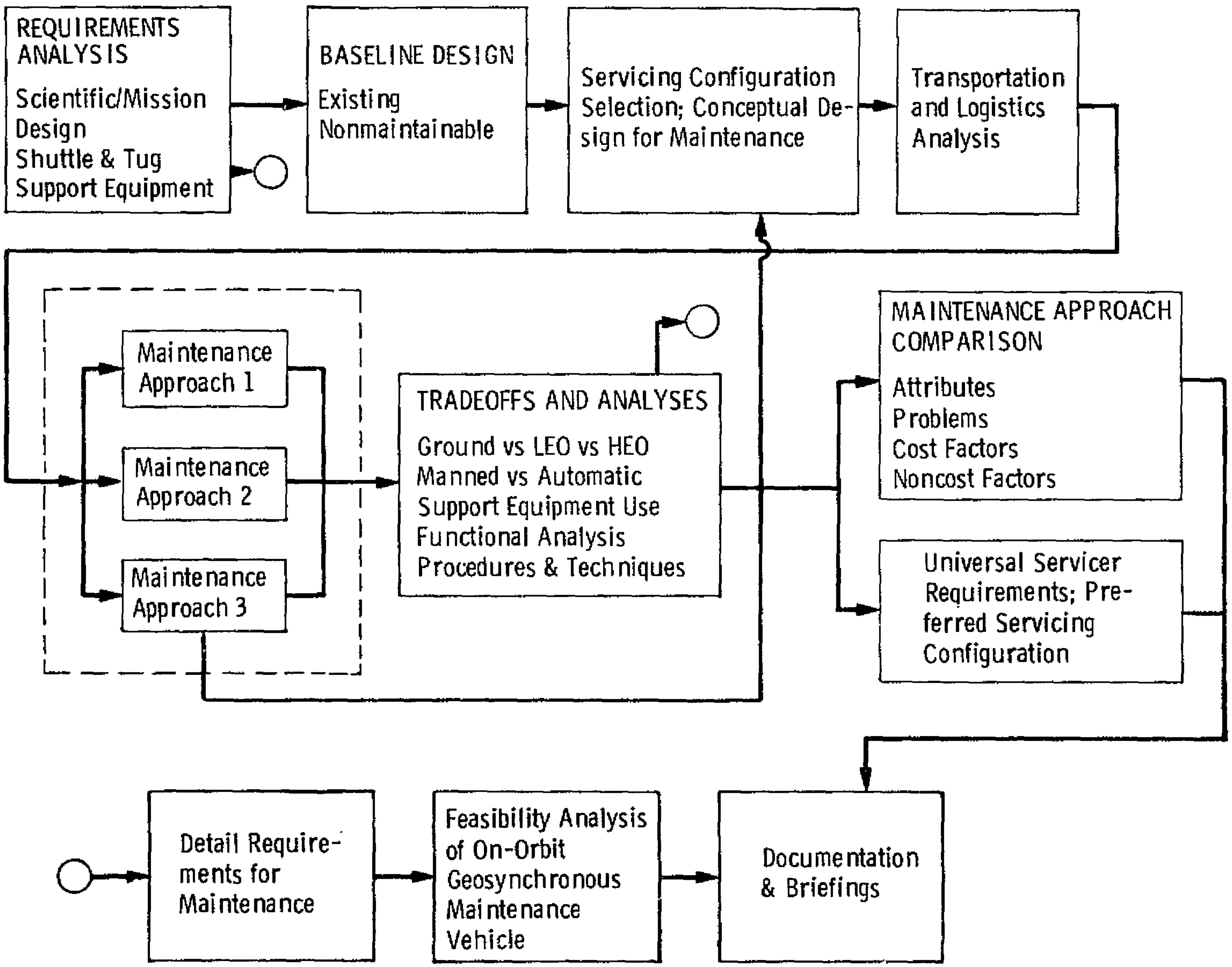
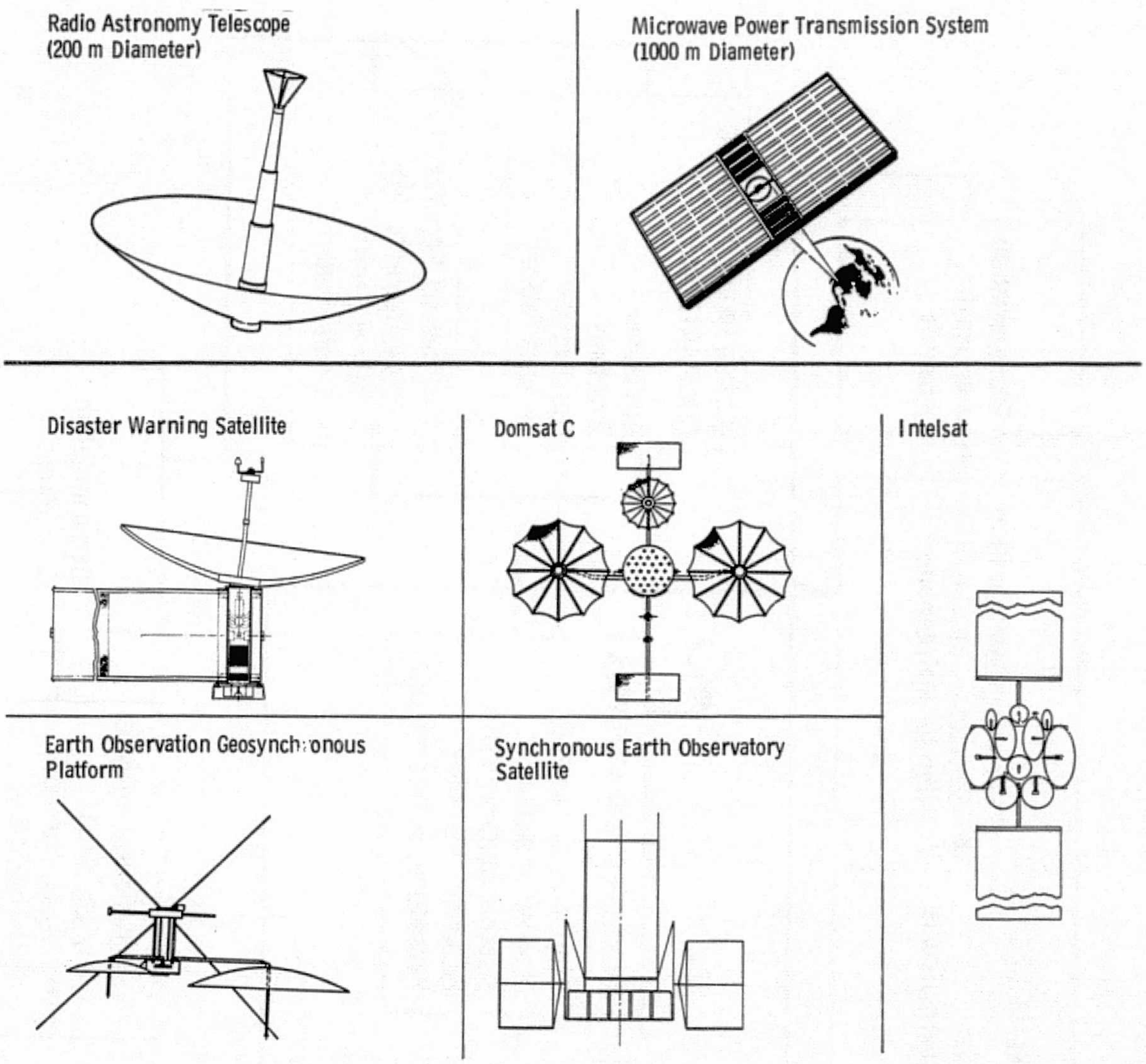


Figure I-2 Study Task Flow - Maintenance of Space Systems


Radio Astronomy Telescope
(200 m Diameter)
Microwave Power Transmission System
(1000 m Diameter)
Disaster Warning Satellite
Domsat C
Intelsat
Earth Observation Geosynchronous
Platform
Synchronous Earth Observatory
Satellite

II. ASSEMBLY OF THE MICROWAVE POWER TRANSMISSION SYSTEM (MPTS)

A. INTRODUCTION

The MPTS is part of the Satellite Solar Power Station (SSPS). The SSPS, shown in Figure IIA-1, will operate in geosynchronous orbit and will convert solar energy into microwave energy, which is beamed to a receiving station on earth. It is then converted back into electrical power for domestic use.

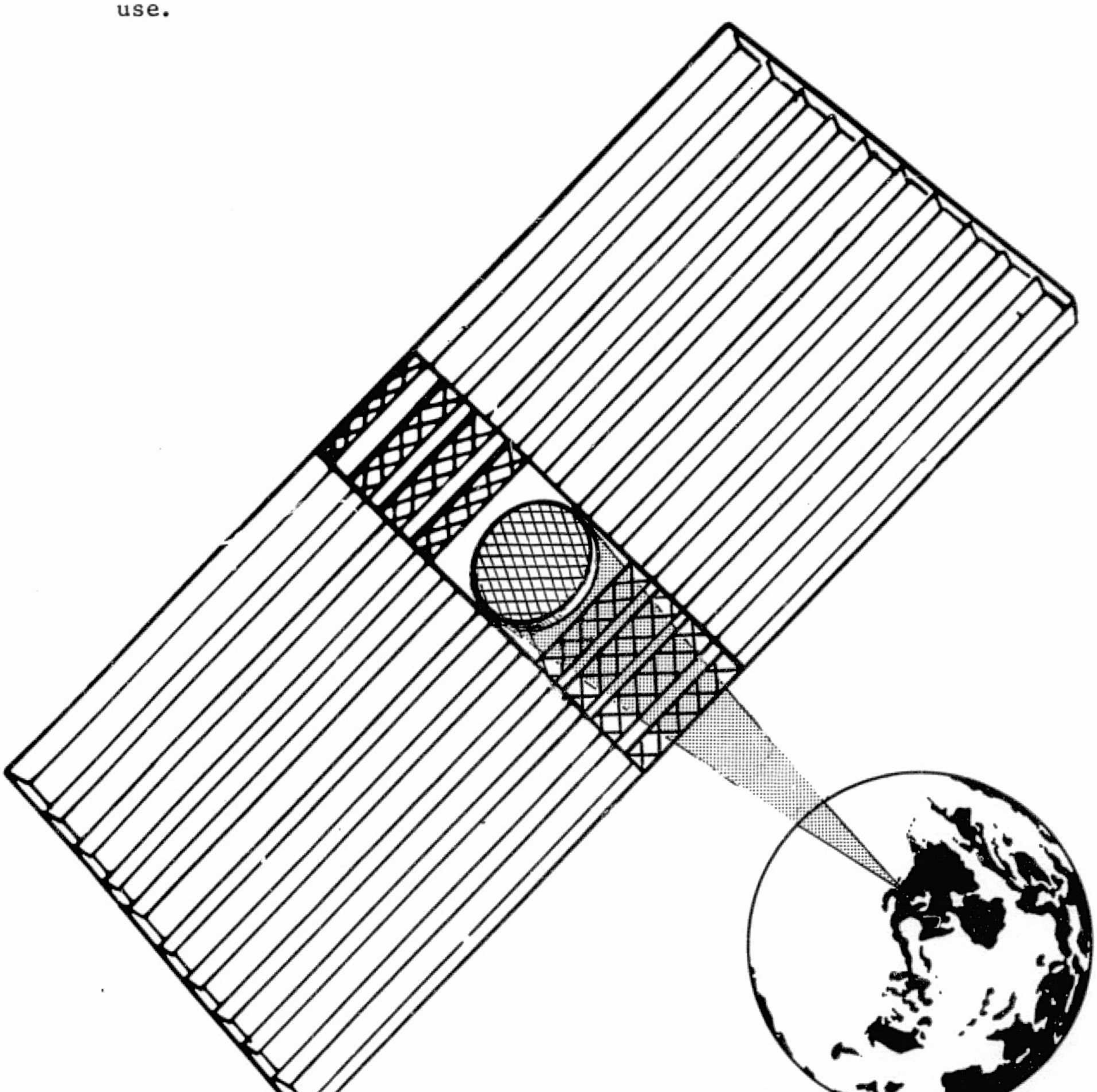


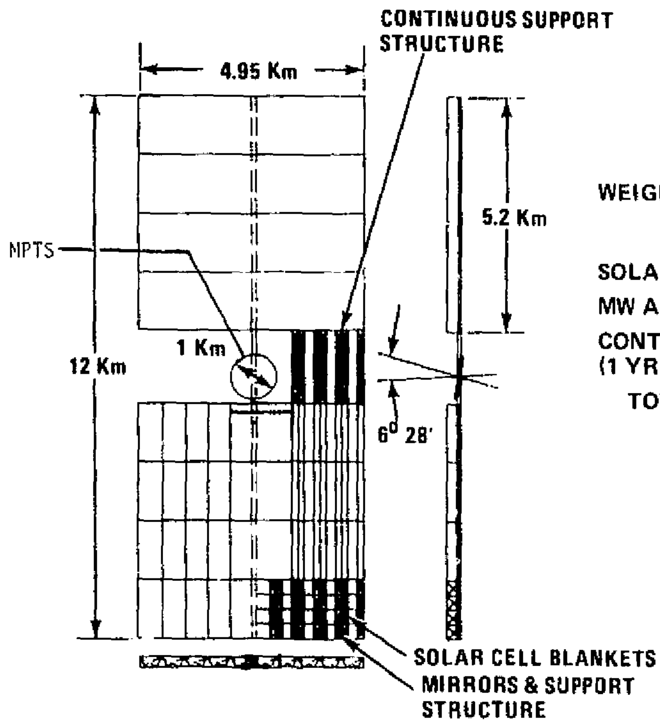
Figure IIA-1 Satellite Solar Power Station

The MPTS is the 1 km diameter "disk" shown at the center of the SSPS in Figure IIA-1.

This chapter discusses the requirements, conceptual design, tradeoffs, procedures and techniques for orbital assembly of the support structure of the MPTS. We begin by describing the Raytheon/Grumman[1] design and design requirements which is used as a baseline or starting point for our redesign for assembly in orbit. In Section B our assembly design is followed by thermal and stress analvses and discussions of the packaging, alignment and subsystems requirements. A discussion of manned vs. automated and transportation tradeoffs are discussed in Section C of this chapter.

B. REQUIREMENTS (TASK 1)

Figure IIB-1 is a drawing of the baseline SSPS showing the 1 km diameter MPTS.



WEIGHTS

	Kg X 10^6	LB X 10^6
SOLAR ARRAY	9.8	21.61
MW ANTENNA	1.66	3.65
CONTROL CONSUMABLES (1 YR)	0.02	0.04
TOTAL	11.48	25.30

GRUMMAN

Figure IIB-1 Baseline SSPS

[1] Contract NAS3-17835

The Raytheon/Grumman design concepts are shown in Figure IIB-2 through IIB-5. Since the microwave generators, waveguide panels, and gimbal structure were not well defined at the time of need, our study was confined to design concepts and assembly techniques for only the antenna support structure.

The microwave (MW) antenna, depicted in Figure IIB-2, is composed of a structural grid to which amplitrons, waveguides, and associated microwave electronics are attached.

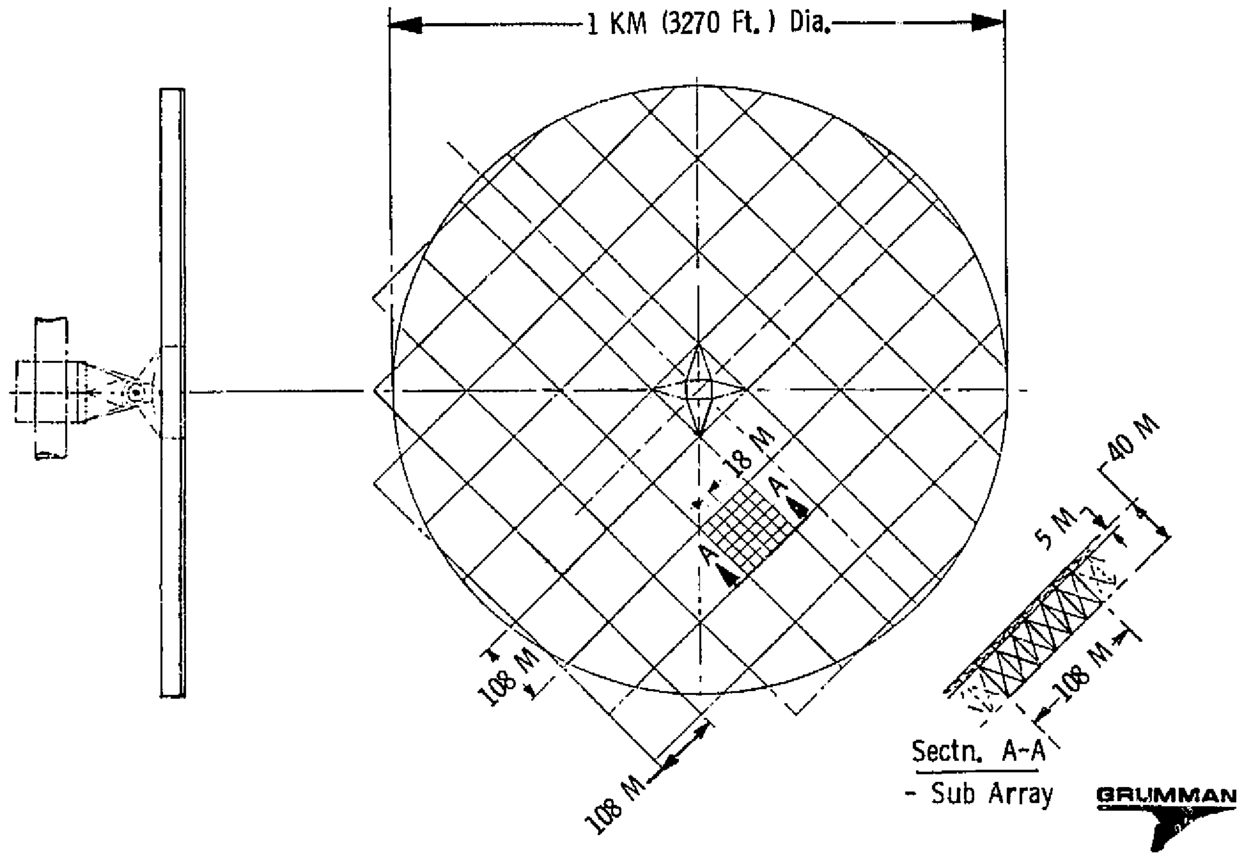


Figure IIB-2 Raytheon/Grumman Antenna Structural Arrangement

The MW antenna rotates on the main structure mast to maintain earth pointing as the solar cell structure maintains sun pointing. Power is transferred to the antenna through the rotating joint. The antenna is pointed in elevation by actuators at the elevation joint. Figure IIB-3 depicts the rotary joints.

The transmitting array at the face of the antenna is broken into 18-meter squares (2,484 in number). Each square section is pointed independently, by use of screw jacks controlled by the pointing circuitry. Pointing accuracy requirements for the transmitting array panels are 1 arc minute.

Figure IIB-4 presents a detailed view of one of the Raytheon/Grumman 108 x 40 m structural elements which make up the 1 km microwave antenna. The base structure is composed of 36 modules approximately 18-m square by 5 m deep.

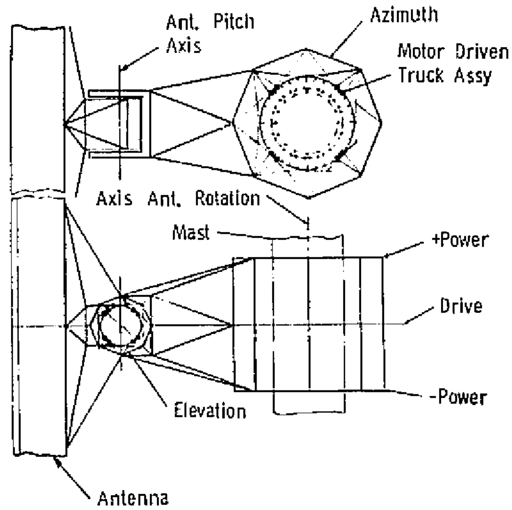


Figure IIB-3 Raytheon/Grumman Two-Axis Antenna Pointing Gimbal

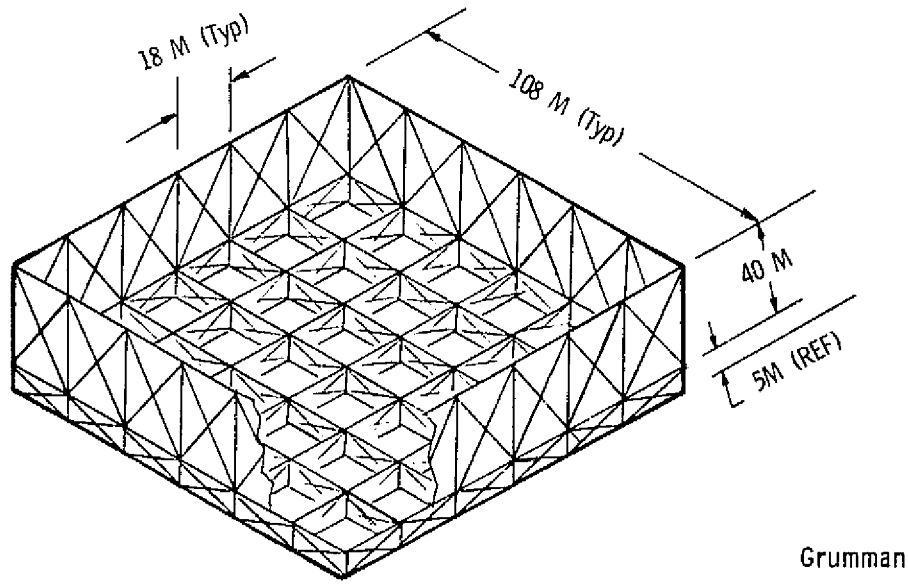


Figure IIB-4 Raytheon/Grumman Typical Antenna Construction Technique

Figure IIB-5 presents a cross section detail of a segment of the Raytheon/Grumman microwave antenna structure. This design uses four basic beam sizes: 18 m (59 ft) x 3 m (9 ft); 35 m (115 ft) x 3 m (9 ft); 18 m (59 ft) x 1 m (3 ft); and 5 m (15 ft) x 1 m (3 ft). Each of the beam segment junctions are supported by tension cables that must be emplaced after the beams are assembled. This drawing shows both the support structure and waveguide array. The following discussions only consider the support structure for space assembly and does not include the waveguide assembly. Other MPTS baseline design factors and requirements will be discussed, where appropriate in the conceptual design section II.C.

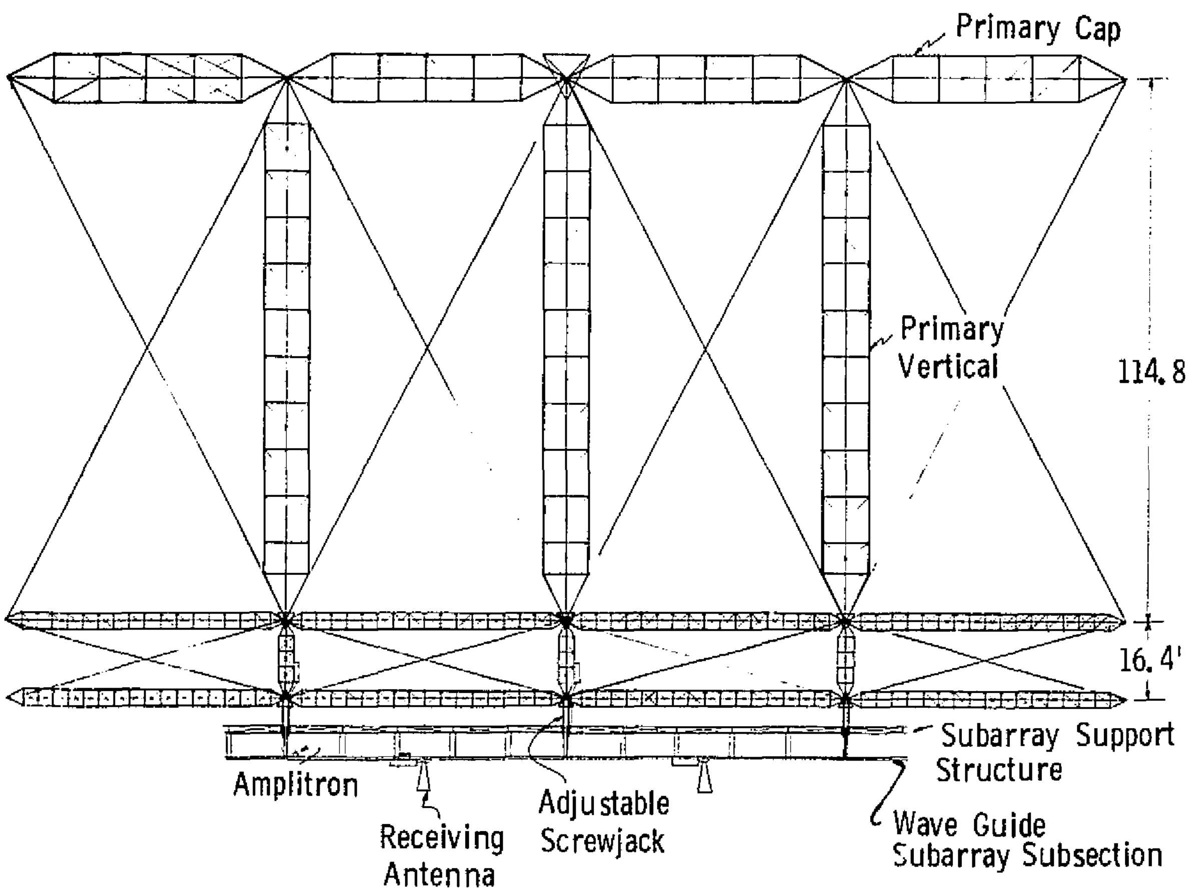


Figure IIB-5 Raytheon/Grumman Antenna Cross Section

C. CONCEPTUAL DESIGN (TASK 2)

This section consists of 4 parts: (1) Structure and Mechanisms, (2) Packaging for Delivery to Orbit, (3) Structural Alignment Concept and (4) Assembly Support Subsystems. Emphasis is placed here on design concepts with only passing reference to manned vs. automated tradeoffs, transportation and assembly procedures. These latter subjects are treated in more detail in Section D and E of this chapter.

1. Structure and Mechanisms

This part is divided into 4 sub-parts: (a) Structural Configuration, (b) Raytheon, Grumman/Martin Marietta Structural Comparison, (c) Mobile Assembler and (d) Stress and Thermal Analysis.

a. Structural Configuration - The objective of this task is to conceive and design an assembly technique for the MPTS microwave antenna support structure. Initially, the Raytheon/Grumman (GAC) structural design and several other concepts were reviewed and their proposed assembly procedures analyzed. We found that these structures and their assembly procedures were not designed for easy assembly in orbit and not totally compatible with the presently defined Space Transportation System (STS). We, therefore, chose to redesign the support structure and develop a detailed assembly technique. (In Section 2, we discussed the Raytheon/GAC approach). Our guidelines were:

(1) Compatible with the STS
(2) Building block approach
(3) Adjustable joints
(4) Universal usage

To understand some of the problems involved in orbital assembly of large structures, one can start by considering the "assembly" of large buildings on earth. These structures are made up of prefabricated members that are transported to the building site and temporarily stored. The building crews position the members through the use of large cranes one floor at a time. Ironworkers are available to align hole patterns prior to the installation of fasteners at each joint. Each level of structure is accurately aligned with surveying equipment. Members are braced at this point and final fastener installation is made before proceeding with the next level of construction.

When this procedure is translated into orbital structural assembly, it is obvious that an orbiting structure must compare structurally in many ways with an earth structure regardless of the means used to accomplish the end product. It must be (1) assembled rigidly at each joint so the total structure will not exhibit looseness when completed, and (2) alignment must be maintained within some predetermined tolerance throughout the assembly period. There are many shapes and types of structural concepts which may be considered for orbital structures, but in any case, these two requirements must be satisfied. For earth assembled structures, alignment and rigidity entail a small part of the task at hand. Many people are available to properly install fasteners, operate cranes, check structural alignment and make the necessary corrections. On structures as large as proposed for an orbiting power station, for instance, the support of such large numbers of workers for long periods of time in the space environment does not appear practical. However, we anticipate the use of man in space for LEO alignment tasks during initial assembly, for repairs during the inevitable failures of assembly equipment, and for maintenance during the operational phase. This philosophy is the basis for the MMC design of large support structures.

Figure IIC-1 is a view of a typical structural section of the total structure. The upper and lower trusses are triangular shaped, constructed from tubular members. Each member is attached to the previous truss member at each of the three legs. By doing this each member is truly continuous, which not only is structurally desirable, but also simplifies the joint design as well as the design of the total member. Note, that as the trusses cross, their neutral axes do not intersect. Earth structures would not be designed this way as this occupies too much space. For the space application, assembled volume is not a constraint. The minor eccentricities of this arrangement have minimal structural effect. The upper and lower trusses are tied together with similarly constructed but square-shaped columns. Each leg of the column intersects the centerline of the two crossing legs of the intersecting triangular truss members.

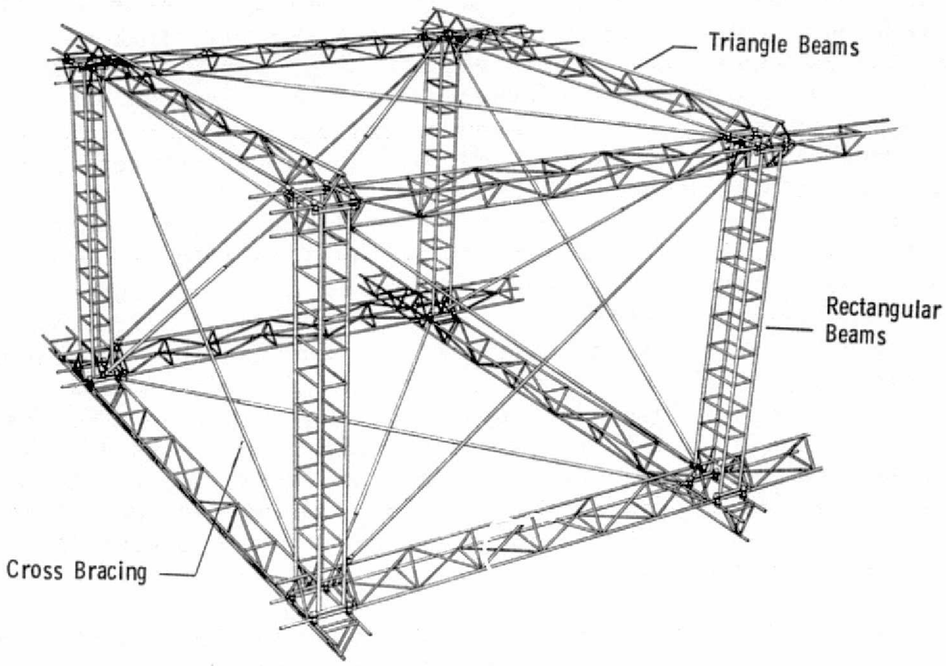


Figure IIC-1 Typical Structural Section

The connection at this intersection is an important part of this design. Figure IIC-2 shows a closeup view of how the members are fastened. Flat surfaces are utilized as the common member interface with a form of welding being used as the fastening technique. The purpose of the flat surface interface is to allow the members to be shifted for final alignment prior to final fastening. In a few cases, a pin at one end of a truss will be used for initial positioning. In most cases, the three leg attachment of the continuous member will adequately locate one end of the truss.

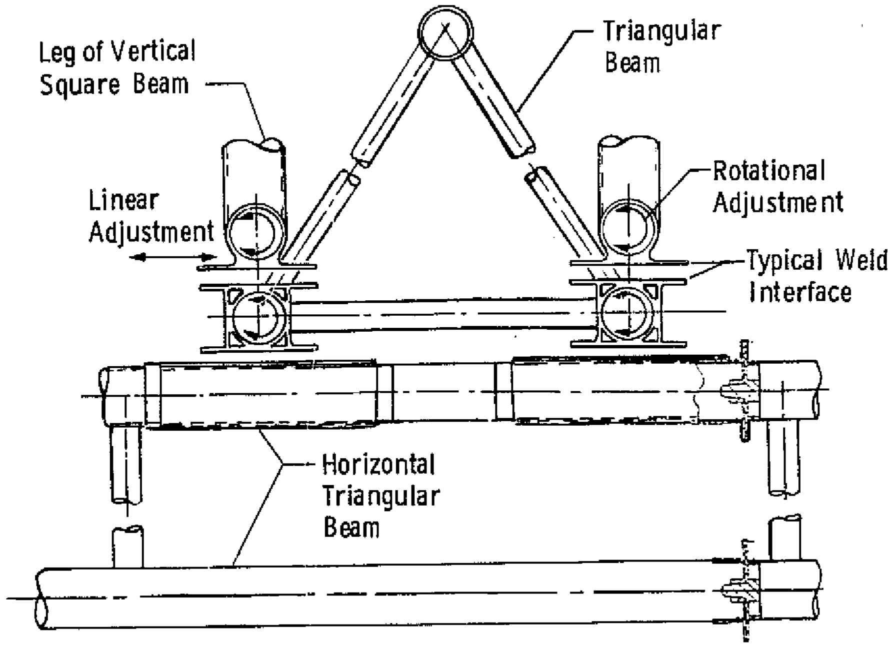


Figure IIC-2 Structural Joint Details

Telescoping tubular tension members (cross bracing) are used to stabilize the structure in all planes.

Figure IIC-3 shows the entire antenna support structure, consisting of 2709 cubes.

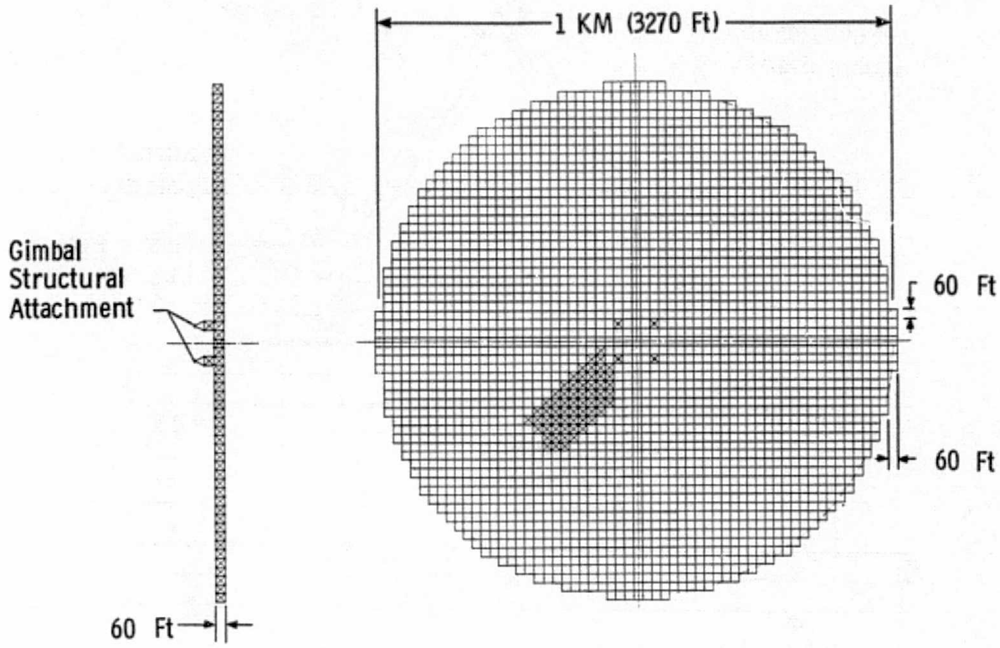


Figure IIC-3 Total Microwave Antenna Support Structure

Table IIC-1 gives the approximate weight statement.

Table IIC-1 Microwave Antenna Support Structure Weight Statement

Member Quantity	Member Type	Weight per Member	Total Weight
11,056	Triangle Members	91 lbs	1,006,096 lbs
2,820	Square Members	101 lbs	284,820 lbs
21,884	"X" Braces	30 lbs	656,520 lbs
TOTAL 32,700			1,947,436 lbs

It appears feasible to perform many of the structural assembly tasks by a mobile assembler, as shown in Figure IIC-4. These tasks would include transporting beams over long distances, installation of the beams, and positioning the beams. The mobile assembler is discussed more fully in Section 3.

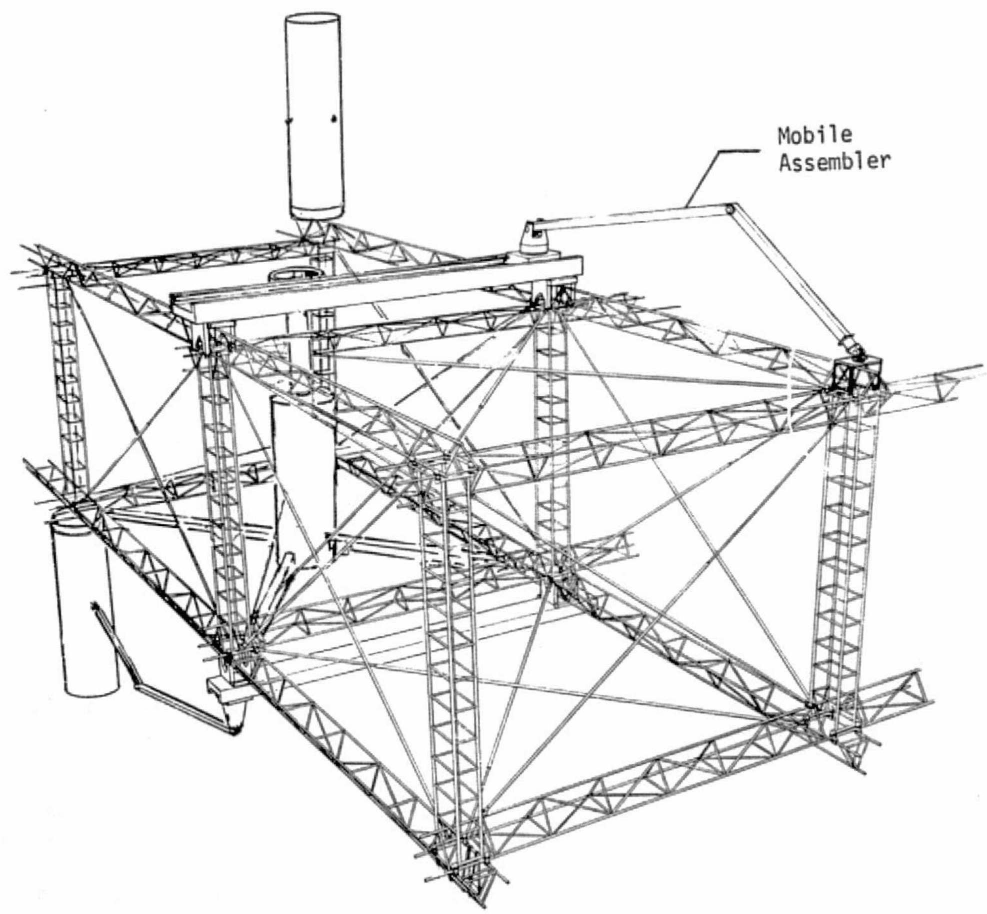


Figure IIC-4 Assembly With Mobile Assembler

The central core system concept was conceived to initiate the assembly process and to provide the necessary system support for continued operation of the construction operation. Figure IIC-5 is a general arrangement drawing showing the primary items which make up this assembly.

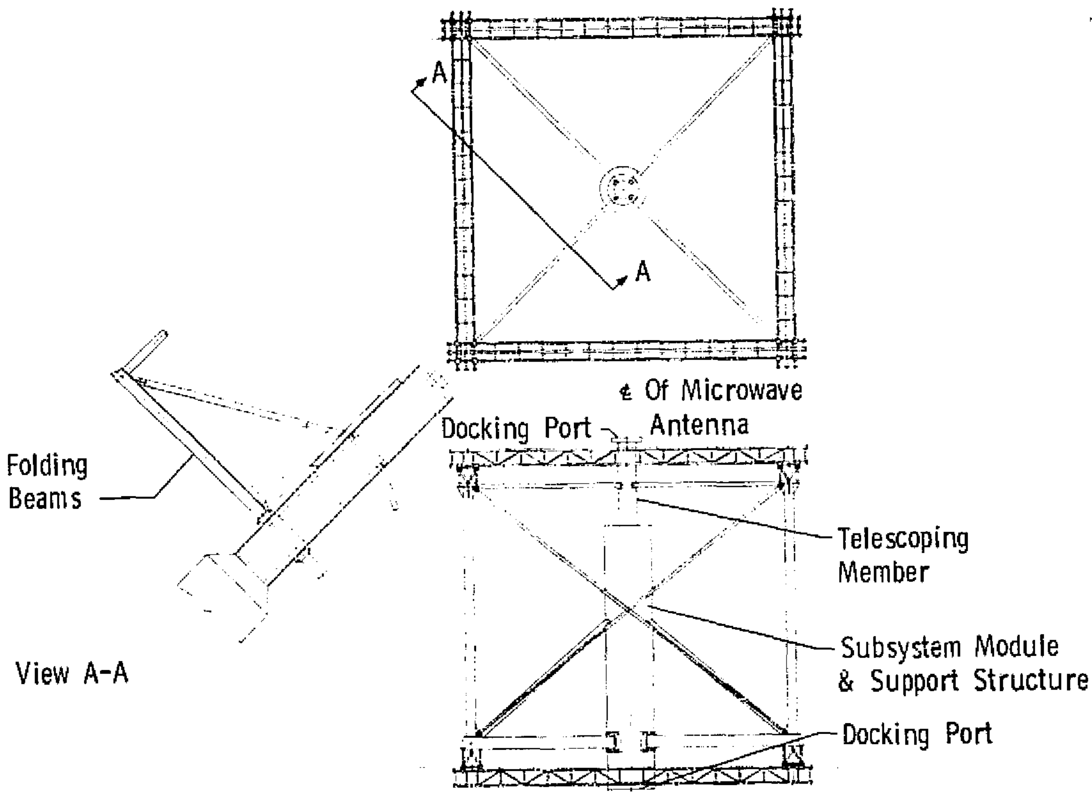


Figure IIC-5 Core Structure Assembly

This part of the structure is delivered to Shuttle orbit by one Shuttle mission. The 8-foot diameter central core cylinder will support the four folding box beams and braces and form an alignment fixture to start the first square assembly. The Shuttle remote manipulating arms will be used to deploy this assembly and to place the remaining members necessary for the first 60-foot structural cube. Mobile assemblers will be placed on both faces of the completed core structure and the structure will be deployed from the orbiter. This first 60-foot cube will actually be an active orbital vehicle with all systems necessary for self-support. It will have electrical power system solar panels, attitude control and a complete communications system. At this point, the assembler can be directed from the ground or Shuttle.

Techniques for fastening structural elements in an orbital assembly must be optimized for the mission and design concept. Reliability, safety, and feasibility are the primary criteria for choosing fastening techniques. However, weight, volume, repetitive manned activity, and time should be minimized. Eligible fastening methods can be placed in three general categories: mechanical, chemical, and metallurgical.

Mechanical fastenings, such as standard bolts, rivets, etc., require extensive use of man and compensation for tolerance buildup (redrilling, etc.). Mechanical couplings require precise alignment to prevent binding during insertion and they do not provide rigid attachment.

Chemical bondings are less efficient than metallurgical bondings and therefore require greater structure to provide more faying surface. Mixing and applying two-part bonders, waiting for setting, and outgassing present other problems.

Metallurgical bonding processes, which include explosive, friction, and fusion welding, appear to be the most feasible fastening techniques for assembly in orbit. Welding processes, in general, have the disadvantage of not being separable in the event that readjustment is later needed. For welding processes, support equipment becomes a major consideration. Means must be provided to hold the joints in contact and apply the welding energy source. Resistance welding (spot) is the most simply applied process, requiring only the supply of a high electrical current (300,000 amps for 0.5 cm material). In space, this short duration (0.5 second) current would most probably be

supplied by capacitors. However, this method would require heavy conductors. Fusion welding requires the formation of a puddle of molten metal between two structural parts by a concentrated heat source. Simple arc welding will not apply since the arc is a plasma column of an inert gas and the gas cell will not be sustained in space. Electron beam applies well because of the advantage of the ready-made vacuum. However, this method requires periodic maintenance, calibration, and adjustment. Assembly in space offers a possibility of using solar flux, in concentrated form, as a heat source. The solar energy can be conducted to the weld site through heat pipes or in direct focusing.

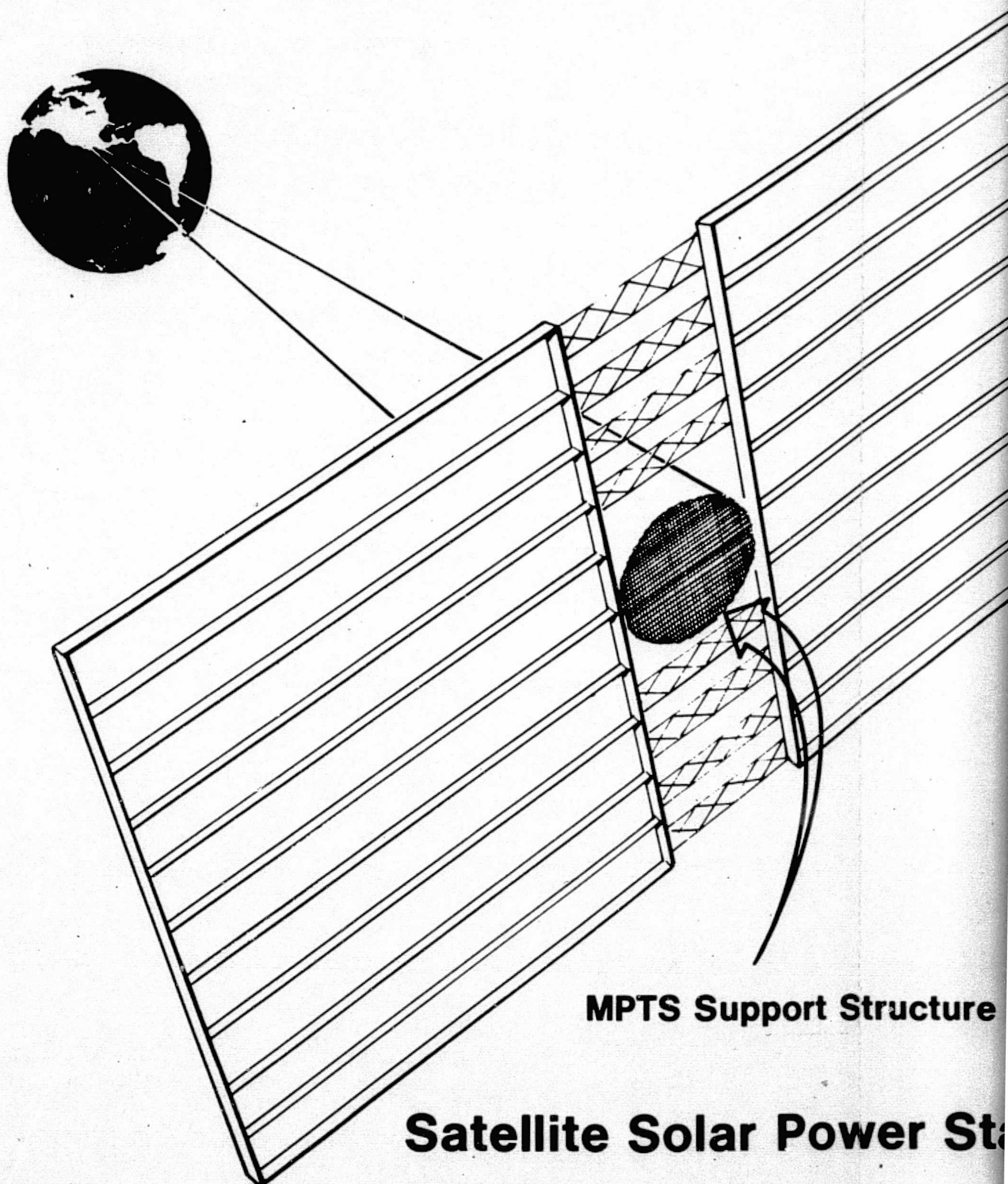
Thermite-type fusion welding, electrically activated through the manipulators, has been assumed as the fastening technique for the support structure in this study. More investigations and analyses should be conducted to define fastening techniques in greater detail.

Figure IIC-6 is an artist's concept of the MPTS support structure in the final phases of construction.

b. Raytheon, Grumman/Martin Marietta Structural Comparison - During the initial stages of this study, a review was made of the Raytheon/GAC MPTS antenna support structure design. The purpose was to identify assembly procedures and techniques and the structural characteristics of the design. From the operational standpoint, the structure appears to be more than adequate to resist deflections due to gravity gradient imposed loads. A good understanding of the thermal distortion and stress problems was shown.

As this study proceeded into the more detailed aspects of orbital structural construction, the design of member joints, alignment and member handling began to impose more of an impact on member design and construction techniques. For example, it is important that as members are joined together, the attachment be rigid, otherwise the desire for a total rigid structure is lost. Also, some reasonable structural plane flatness should be maintained which means that the member attachment joints should be designed with adjustment and alignment instruments employed to indicate proper alignment prior to final attachment of the member. The members should be of a reasonable length to facilitate handling.

Our analysis determined that some modification to the GAC structure was necessary to realize a more workable fabrication and assembly technique.



Satellite Solar Power St

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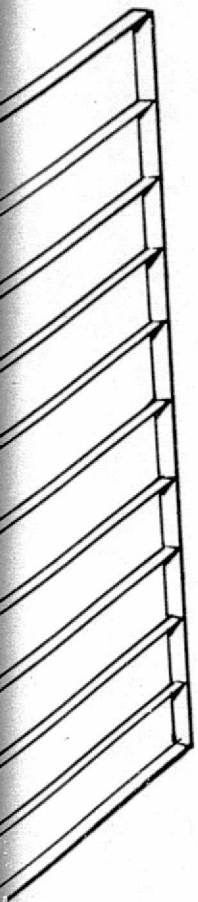
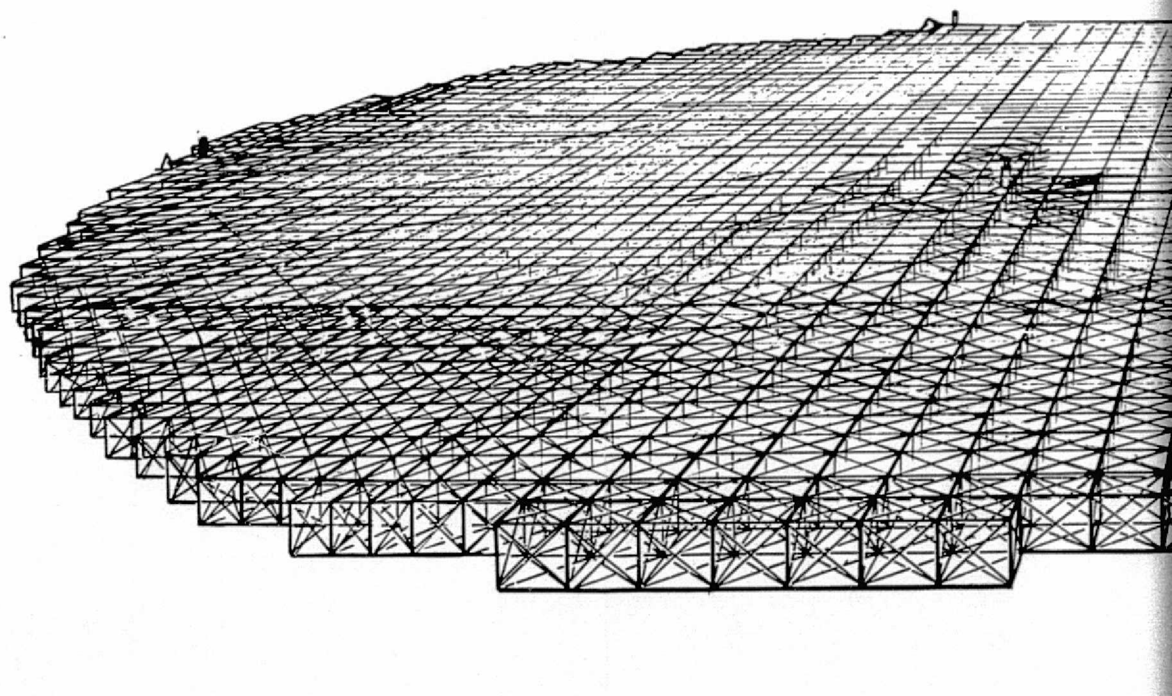


MPTS Support Structure C



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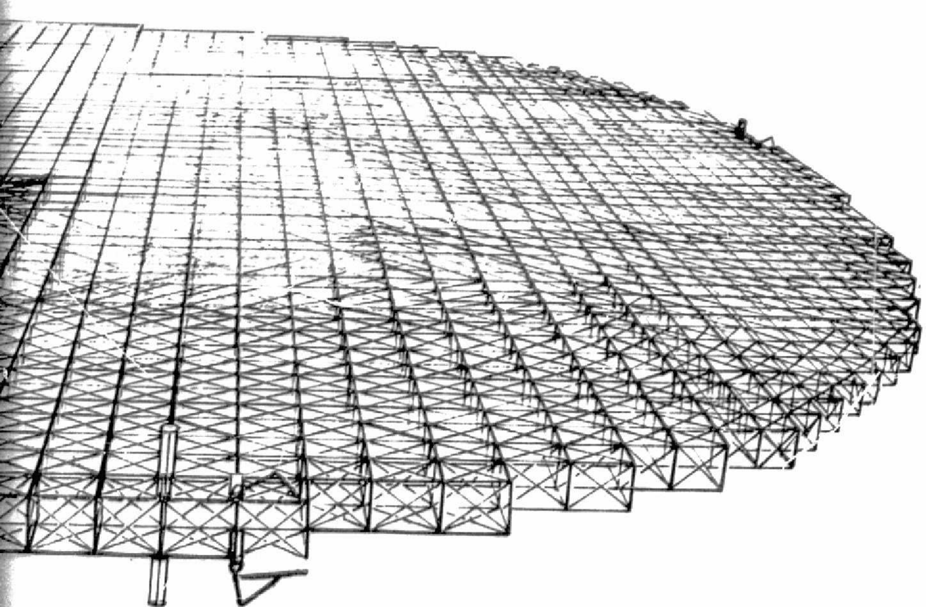
FOLDOUT FRAME 2

e Overview



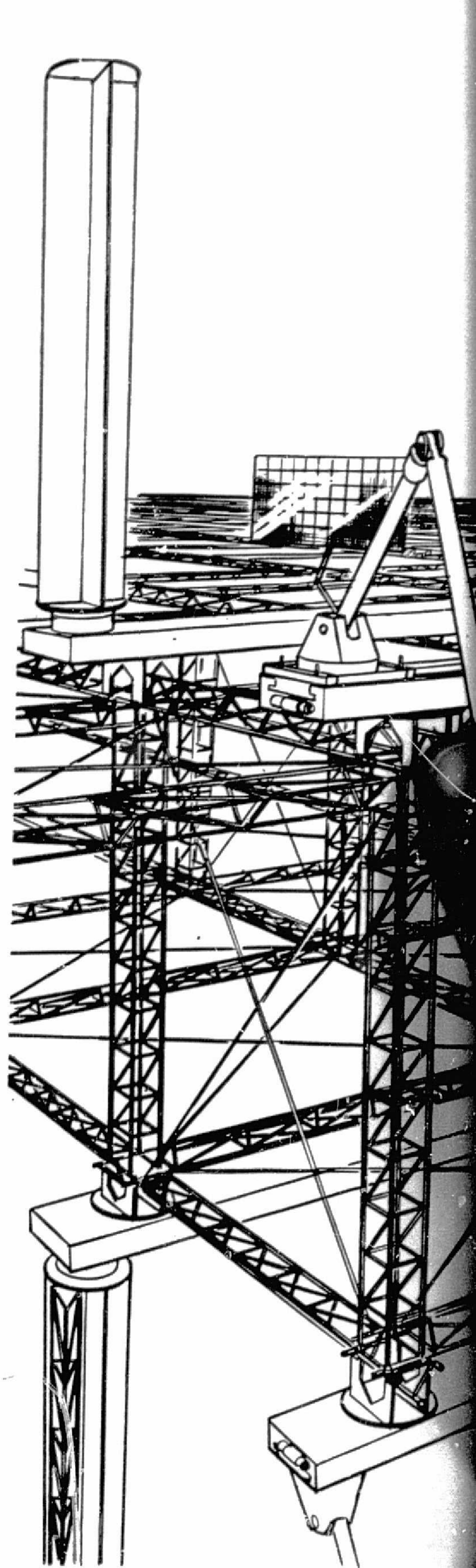


Figure IIC-6 Artist's Concept of MPTS Support Structure Nearing Compl

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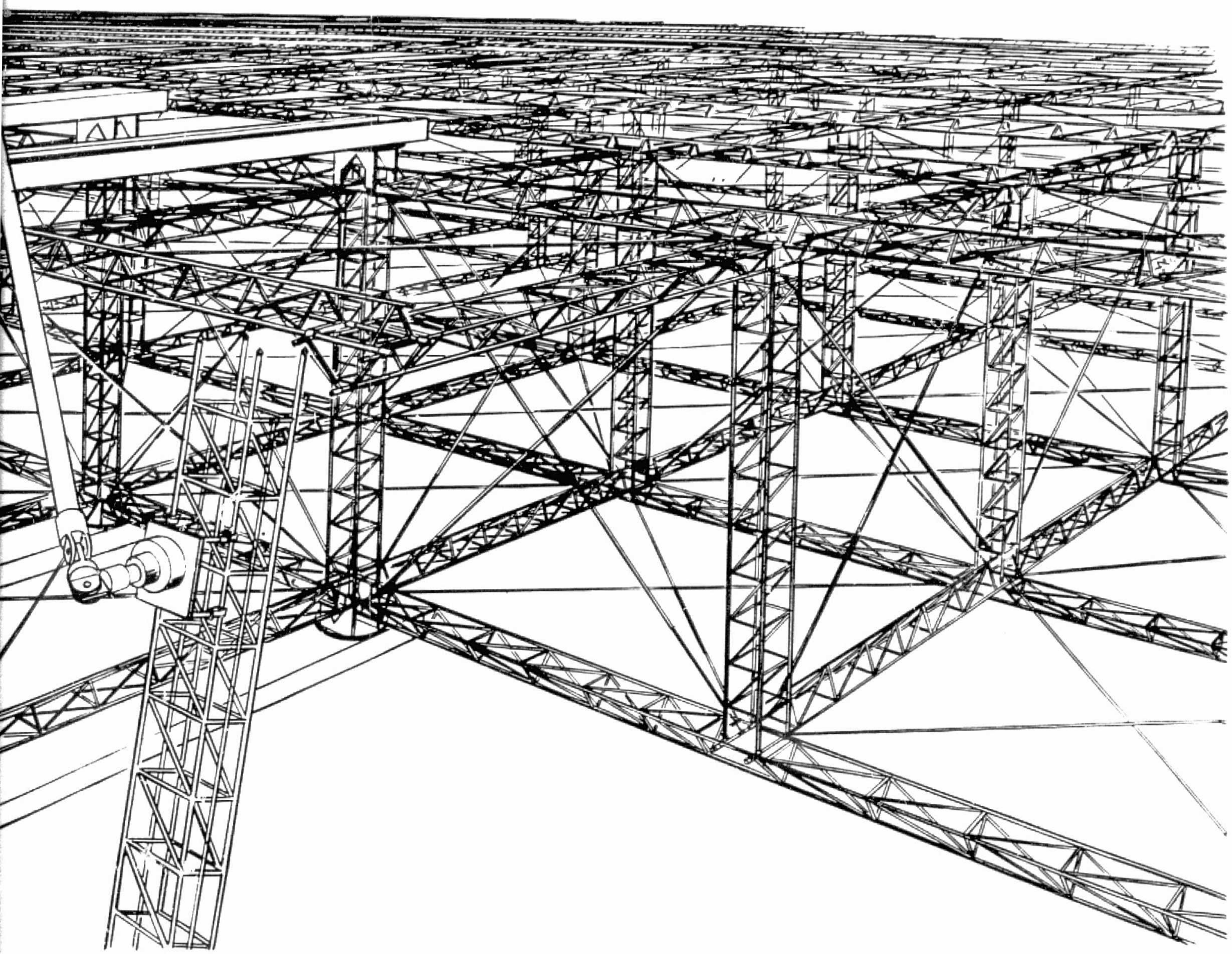
MPTS Support Structure in Assembly Phase



In order to establish a design philosophy, it is important that a few basic considerations be examined so that the final concept will not exceed practical limits. These considerations should be:

1) The structural concept developed should be universally applicable, if possible, to many large space structures.

2) Due to the proposed size of these structures, all aspects of beam design, joint design, and assembly methods must be of the simplest form.

3) The use of man in EVA tasks should be limited to the area of maintenance and corrective action type tasks, if possible. The actual assembly tasks are repetitive and can be assigned to semi-automated machine activity controlled from the ground.

Our analysis indicated that the GAC orbital beam fabrication machine imposed a complexity level on the orbital work effort, that exceeded our desire for simplicity. Our approach utilizes prefabricated foldable members.

This structural member fabrication technique limits member lengths to slightly less than 60 feet to accommodate the cargo bay length. As a result, the GAC structure required considerable modification. Analysis shows that to maintain a given structural stiffness, the GAC 35-meter primary structure plus the 5-meter secondary structure could be reduced to one 18-meter structure, providing the primary structural members were repeated in every bay rather than every sixth bay. Our approach eliminated the double secondary structure, as shown in Figure IIC-7. This modification reduced the total number of members as well as the weight per member, while not sacrificing the structural integrity. Cable cross-bracing was replaced by telescoping rigid tubular members for the sake of assembly simplicity. Stringing and adjusting cables would be particularly difficult when considering the lengths and quantity proposed by GAC.

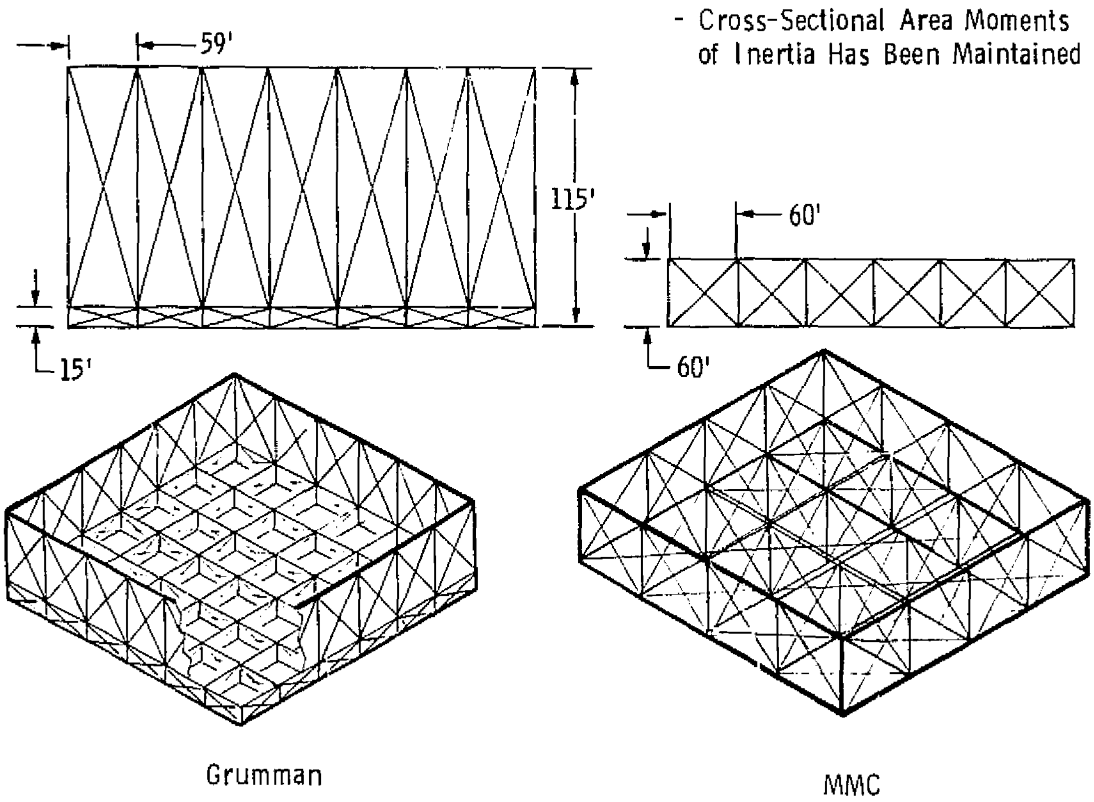


Figure IIC-7 Grumman/MMC Structural Comparison

We recommend the use of an adjustable, welded member joint attachment, rather than the slip joint ball lock type proposed by GAC, (see Figure IIC-8). To assemble a tube within a tube appears to be difficult, considering 115-ft beam lengths, unless considerable gap is allowed between the I.D. and O.D. of the respective tubes. This coupled with the ball lock device would result in a very loose joint. This concept also does not allow for adjustments required for tolerance build-ups and structural alignments.

To summarize, the variations in the GAC/MMC structures are shown in Table IIC-2.

The MMC weight number exceeds the GAC number. The GAC triangular shaped beams are composed of open-cross-section members rather than the round closed-tube members used by MMC which accounts for the additional weight. In the course of our study, additional loading conditions were identified and analyzed over the operational gravity gradient load which sized the Grumman

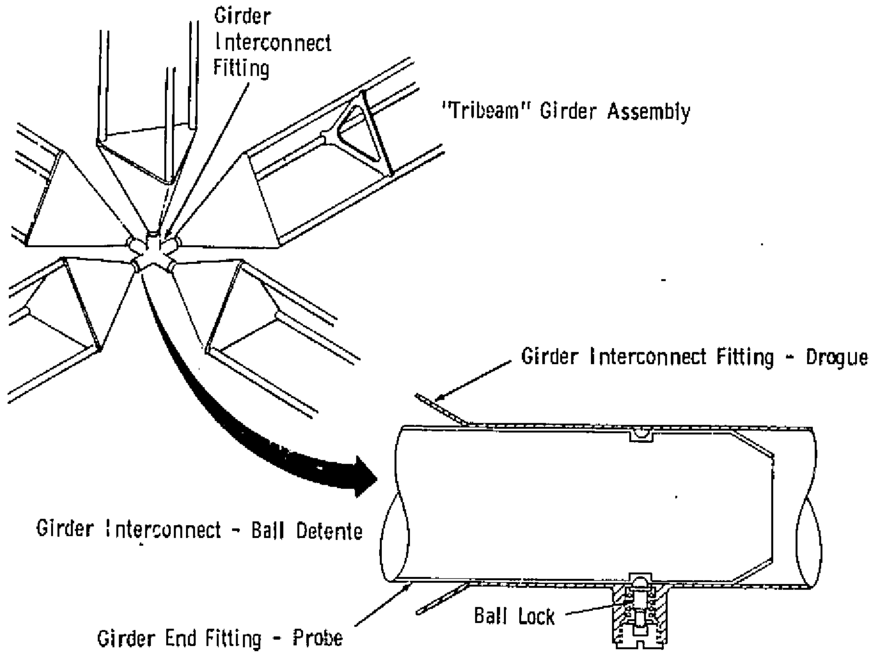


Figure IIC-8 Raytheon/Grumman Structural Joints

structure. Additional loads include, 1) the forces applied by the assembler, as members are placed in position, 2) resupply docking, and 3) thermal stresses experienced during antenna operation. We feel that the Grumman structure would require additional consideration to satisfy these conditions.

Table IIC-2 Summary of Key Antenna Design Changes for Assembly

ITEM	RAYTHEON/GRUMMAN	MMC
• Support structure	Two tier	One Tier
• Support structure depth	131.2 ft	60 ft
• Beam Sizes	(4) 114.8 ft x 9 ft max 16.4 ft x 3 ft min	(1) 56.2 ft x 30 in.
• Adjustment for structural tolerance buildup	None	Adjustable beam intersections
• Compatibility with Shuttle	Beams must be assembled in space	All beams less than 60 ft long and collapsible

The MMC structure consists of two types of main beam members and cross-braces for a total of three discrete types of members. The two beam members have common crossing interfaces which are flat surfaces. This allows angular and side movement to accommodate the continual alignment required as each cube is constructed. These flat contact surfaces will contain pre-prepared welding material that can be activated on command, after the member is placed within its alignment tolerance. Each cross-brace is a telescoping tube which can be final locked on command, as alignment is obtained.

This structure is approximately 60 feet thick and is a continuous array of 60-foot squares, as shown in Figure IIC-9. This building block approach can be constructed to any shape desired, over the plane, such as square or essentially round (microwave antenna), triangular or rectangular. Our stress analysis shows that the microwave antenna support structure thickness can possibly be reduced from 60 feet. This concept could be applied to the solar array support structure.

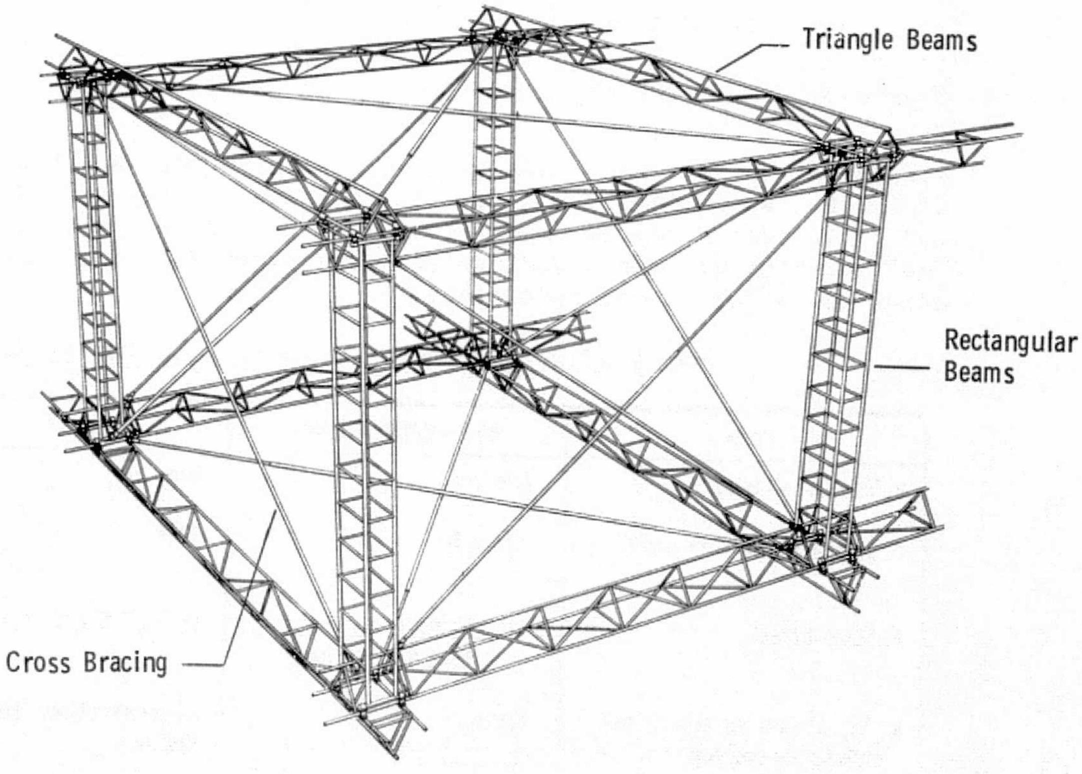


Figure IIC-9 Typical 60-foot Cube

c. Mobile Assembler - A unique Mobile Assembler (MA) was conceived during this study. This assembler, shown in Figure IIC-10. is remotely controlled from the earth by man. The MA system consists of a 72-foot, 7 degree-of-freedom manipulator with a grasp type end-effector, a manipulator base, and two mobile carriages. Figures IIC-10 and IIC-11 show the MA. The manipulator base can be positioned anywhere along one mobile carriage. Beam pallets are docked to the other mobile carriage. Support subsystems for the MA are discussed in part 4 below. These include communications, power, alignment cameras, etc.

The manipulator is the prime component of this system. This 72-foot long manipulator has a reach of 70 feet. We are presently analyzing the manipulator requirements. Preliminary analysis indicates a manipulator joint ordering as shoulder yaw, pitch and roll; elbow pitch; and wrist pitch, yaw and roll. The shoulder roll complicates the manipulator control system; however, our assembly analysis indicates that this shoulder roll is probably needed to reach in and around beams and X braces as the assembly progresses. Since the manipulator shoulder structure travels the length, approximately 60 feet, of the mobile base; it allows the manipulator to work on either side of the structure or to reach either beam pallet. The manipulator end-effector is a unique design, configured to handle the triangular and square beams used in the MPTS structure. The end-effector encloses the beam and contacts at four points, thereby, eliminating normal, single point, grasping problems. Two video cameras are located in the end-effector jaws, each pointing 180° apart. These cameras are used by the operator to view the beam end alignment aids while the final emplacement and alignment takes place.

The manipulator mobile carriage is self propelled and has the capability to move along the structure as the assembly progresses. This is accomplished by making the legs at each end of the carriage foldable and pivoting. The legs fold down to clamp to the tubular structure on both ends. By releasing the folding legs on one end and allowing them to retract 90°, the carriage is cantilevered from the opposite end. The opposite end leg assembly rotates the carriage 90° until the open legs are positioned over the opposite corner structure. The open legs are then closed onto the structure. By repeating this process, the carriage can travel over any of the previously assembled structure. Video cameras are located on each end of the mobile assembler base. These cameras are mounted on

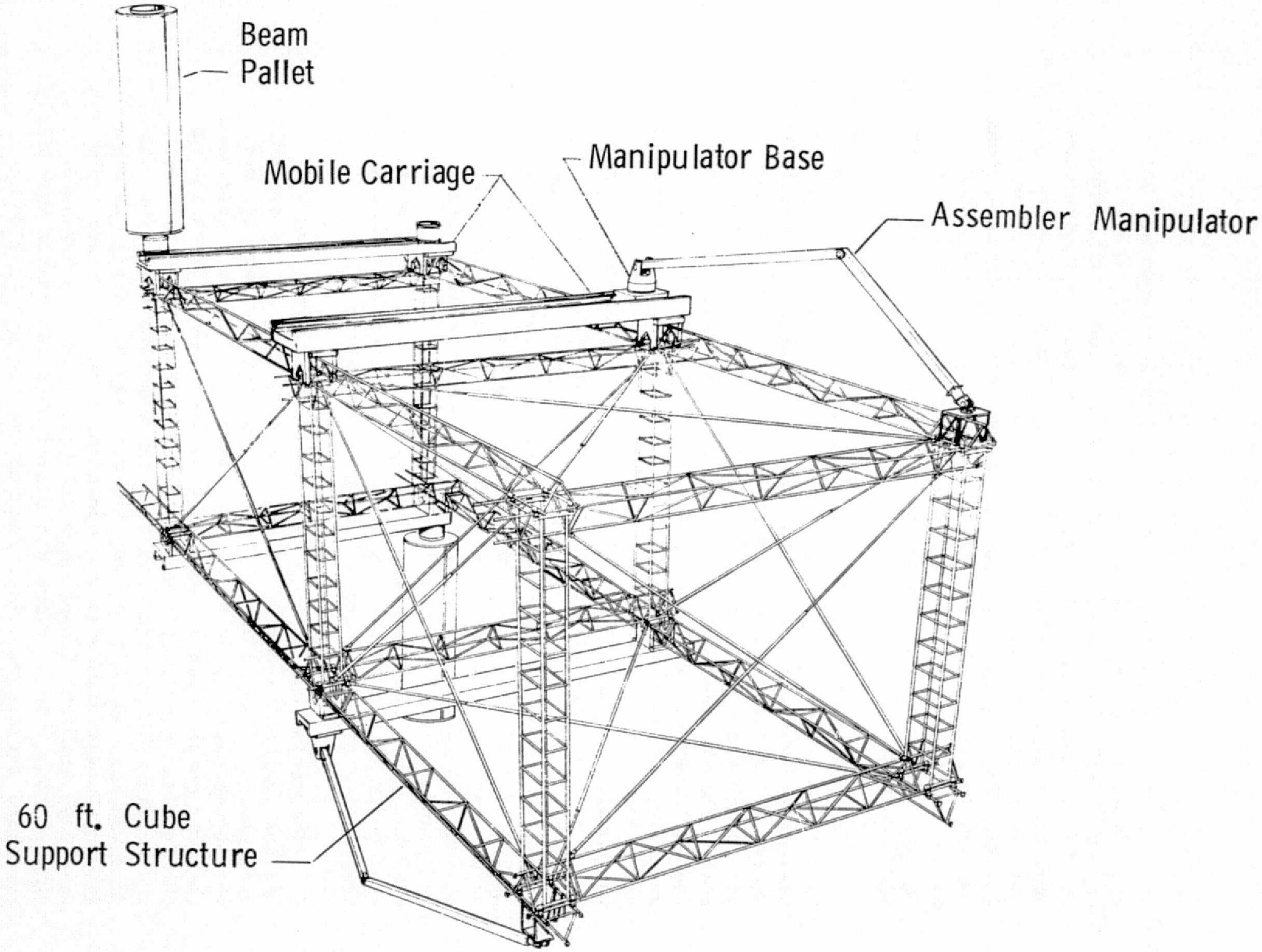


Figure IIC-10 Structure With Mobile Assembler

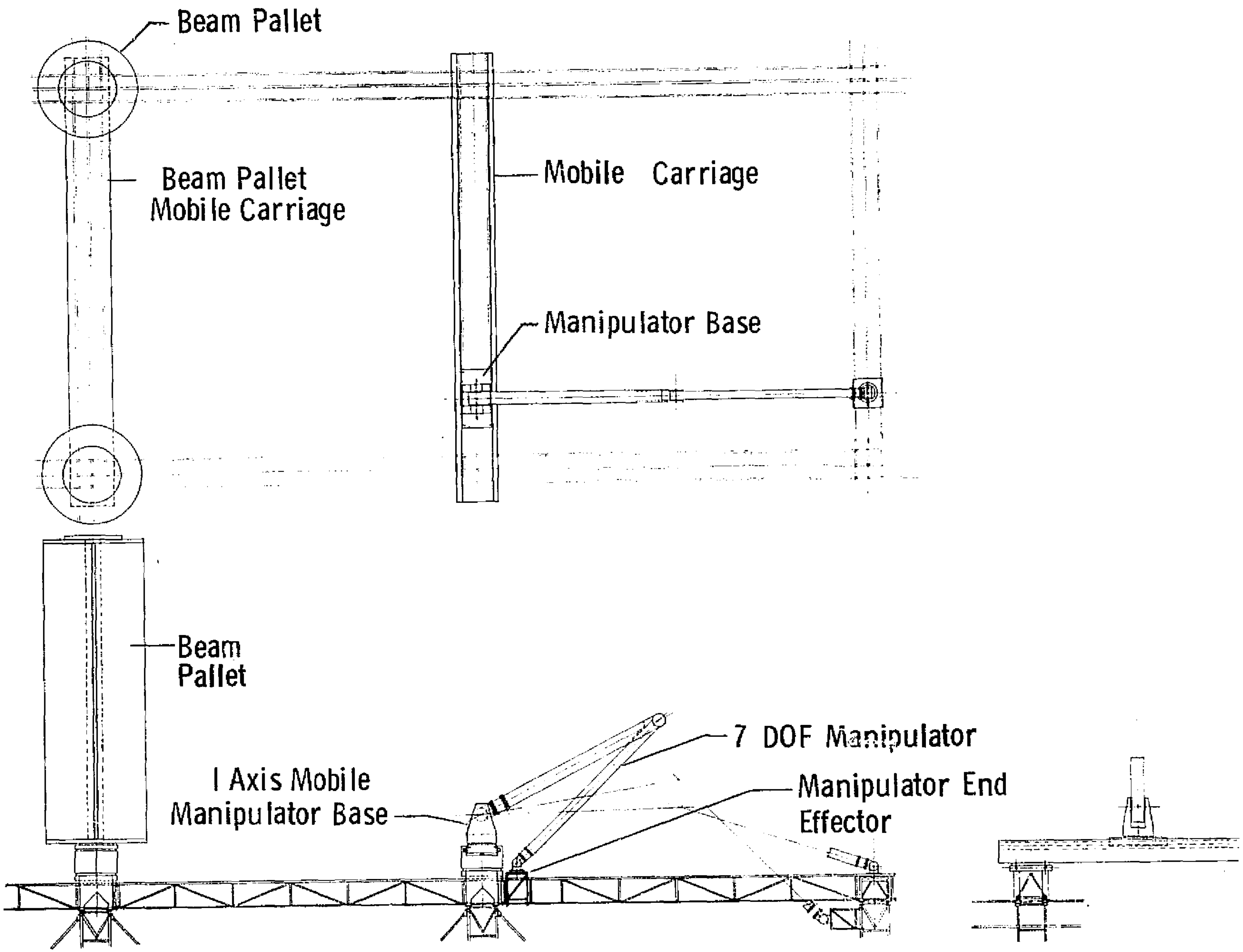


Figure IIC-11 Mobile Assembler and Beam Holder Concept

remote controlled pan/tilt units and have 10 to 100 mm remote controlled zoom lens. These cameras are used primarily for beam translations from the pallet to the assembly site and are also utilized as part of the beam alignment system (discussed in part 4 below). The 38 structural squares to be assembled in LEO will require artificial lighting. The actual requirements are yet to be defined.

The beam pallet mobile carriage is a similar device which has identical operational characteristics to the manipulator mobile carriage. Two docking receptacles are located on the base as shown in Figure IIC-11. This allows a Tug to dock a resupply beam package and then remove an empty canister.

Further study is required to examine ways to utilize the empty beam canisters as part of the MPTS structure. There will be 57 of these 15-foot diameter x 60-foot long canisters that have considerable structural strength. Their use should not be ignored.

The assembly procedure requires a minimum of two sets of mobile assemblers for the 60-foot thick structure since the manipulators cannot easily reach through the structure for assembly on the opposite surface. Obviously, the addition of more assembler sets reduces the assembly time. The assembly procedure is discussed more fully in Section E of this chapter.

d. Thermal and Structural Analysis - This section reports on three specific areas of analysis:

(1) Preliminary stress analysis of the MPTS support structure during the non-operating periods such as load experienced during fabrication and boost. The deflection analysis due to gravity gradient is also reported as cold structure values.

(2) Thermal effects on the completed structure due to antenna radiated heat to the support structure and solar heating.

(3) A second stress analysis based on the high temperature operating conditions.

The study plan for this study does not include a specific task for thermal analysis. On completion of the preliminary stress analysis NASA-JSC suggested that a preliminary thermal evaluation be made due to the interest in this particular structural application. After completion of the thermal analysis, a thermal stress analysis was conducted. This additional analysis revealed a need for new material considerations not included in the first preliminary stress analysis (Item 1 above). Normally Item 1 would be modified to include the new conditions. In this case it was determined that the cold condition analysis should not be altered because of its value in cold structure design. By leaving Item 1 intact, this structural design can possibly be related to the much larger solar panel structure of the microwave power station.

Preliminary (Cold) Stress Analysis - This preliminary loads and sizing analysis was performed on the microwave antenna structure not exposed to the high temperature of the antenna radiating surface. Aluminum is the material chosen for this analysis. Steel and composites were considered in the later analysis.

This analysis will show that a spacing of 60 feet between upper and lower caps is much larger than necessary to maintain stiffness for a structure of this size. However, the analysis looked at loads induced to the support structure alone. The waveguide assembly structure must be analyzed at some later time.

An ultimate factor of safety of 3.0 was used in the analysis. The following four loading conditions were considered:

- o Gravity gradient
- o Orbit transfer - Low earth to intermediate altitude
- o Tug/Pallet orbital docking
- o Assembly manipulating arm loads

The bending moment used to analyze the gravity gradient torque condition was 2000 Newton-Meters as quoted in the Raytheon/Grumman study[1]. The shear and bending moments from this condition was not critical from a strength consideration in the assembled array as shown in Figure IIC-12. A deflection check for this condition showed a rotation of the outer element of the array of .03 minutes. Figure IIC-13 shows the magnitude of these deflections over a range of beam depths.

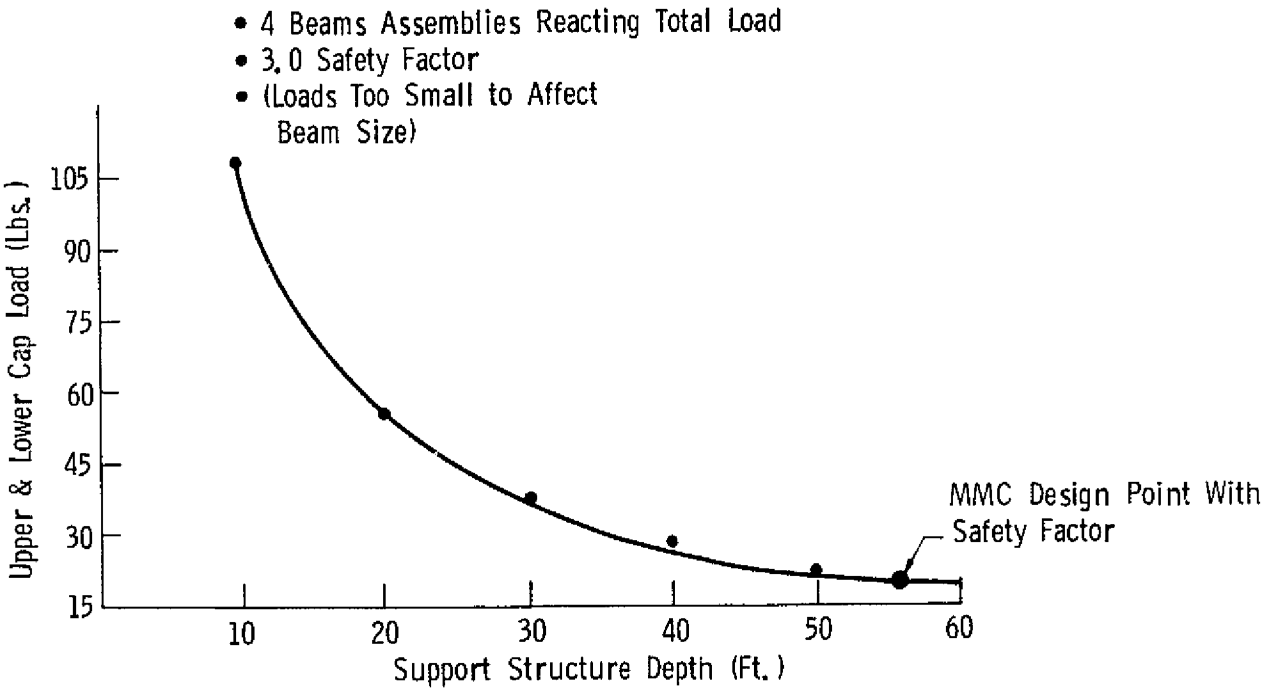


Figure IIC-12 Beam Loads Due to Gravity Gradient Bending

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[1] Contract NAS 3-17835

For boost loads on the MPTS structure, the following was used: two Tugs (in tandem) boosting 38 completed cubes plus a central core from LEO to HEO. This orbit transfer condition gives a limit thrust load of 15,000 lbs. The array of 39 cubical sections has a total weight of approximately 62,000 lbs. The central core section has a center cylinder and supporting truss structure (that interfaces with the orbit change booster) designed to react the 15,000 lbs of thrust load. This condition resulted in the most critical shear load in the bay just outboard of the core section and established the size of the cross struts between the upper and lower triangular column member.

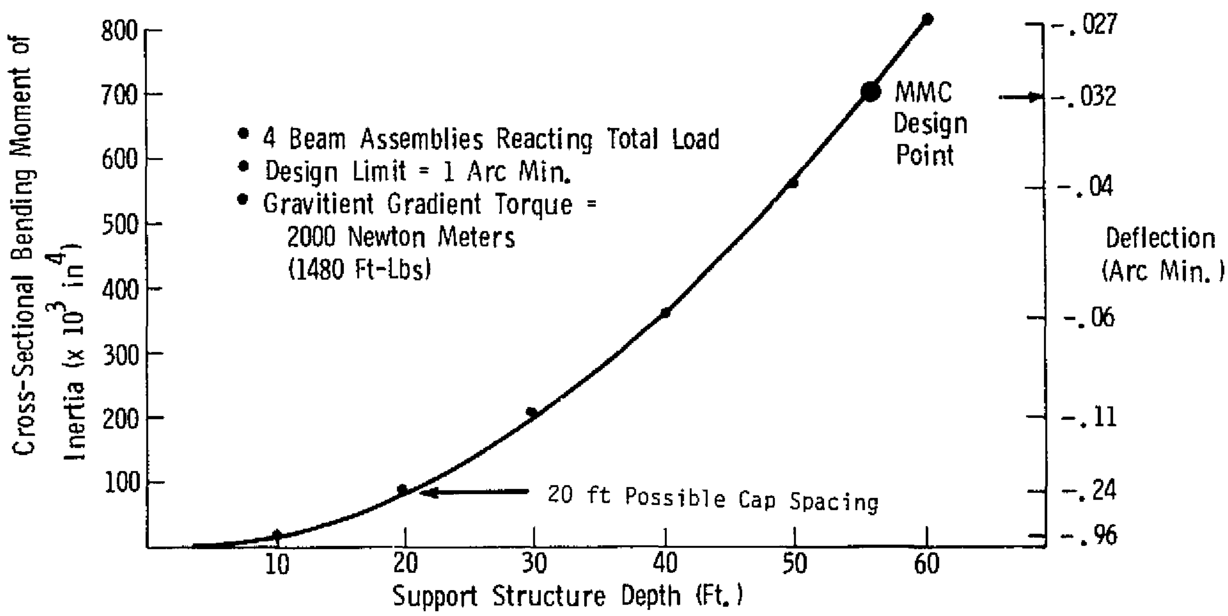


Figure IIC-13 Antenna Deflection Due to Gravity Gradient (Aluminum)

During construction, the Tug will deliver beam pallets to the work site and dock the pallet to the pallet holder. The docking procedure will impose loads on the already assembled structure.

The Tug/Pallet docking condition was based on an assumed minimum docking weight of 30,000 lbs and a velocity of 1.0 ft/sec. The stiffness of the structure was calculated and the

energy of the orbiter was applied resulting in excessive loads and deflections in the structural columns and cross braces. It is recommended that a "soft" docking interface be incorporated that would reduce the level of the docking loads to a level not exceeding the other design conditions. Analysis shows that a spring rate of 30 lbs/in. at the docking interface would result in a maximum force of 581 lbs and a docking mechanism deflection of less than 20 inches. An energy absorber (damper) would be required to limit spring back rate. A constant force system could also be designed to reduce the deflection to less than 10 inches at 581 lbs force if this would be more desirable.

Another loading condition considered in the analysis was the manipulating arm load. A manipulator tip force of 30 lbs (max.) is our present design requirement. This force will normally be exerted 70 feet from the manipulator base. This would enable the cantilevered manipulator column to be deflected approximately 3 inches at the tip under maximum load conditions.

The force applied laterally to the end column, before supporting struts are attached, produces a bending moment in the column of (30 lbs) x (681 inches) = 20,430 in.-lbs. limit. (61,300 in-lbs ultimate).

The material initially selected for the structural elements was 6061-T6 aluminum. The material has good resistance to stress corrosion and is readily weldable. This material should present no procurement problems at a reasonable cost. Steel tubing was later recommended for thermal reasons discussed below.

The columns were checked for overall stability and local stability of each element. The curves in Figure IIC-14 were developed to aid in determining the required member spacing and the member length requirements. The members were limited to 12,000 P.S.I. maximum stress level to prevent yielding at welded joints.

Thermal Analysis - The orbital structural concept presented here is applicable to many space structures; however, the concept was developed specifically for the MPTS. Thermally, the microwave antenna becomes a unique application due to the very high temperatures being radiated from the transmitting surface and the high degree of pointing accuracy required. By making a preliminary thermal and resultant thermal stress analysis of this

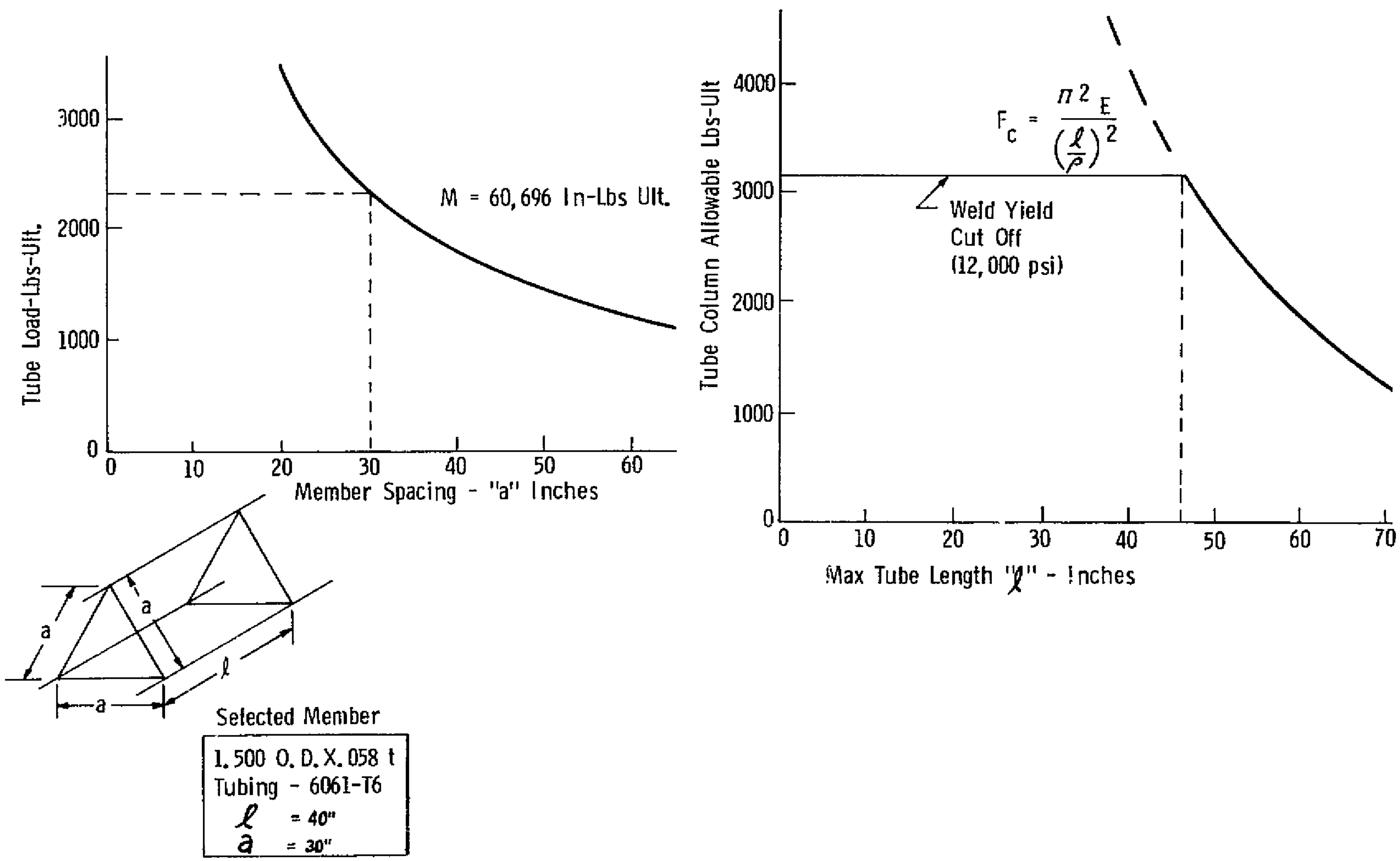


Figure IIC-14 Beam Size Analysis

extreme microwave antenna application, it was felt that a better understanding of all orbiting structural applications was possible. Preliminary thermal analyses concentrated on determining the temperature and the temperature gradients in tubular members of the antenna truss as a function of distance from the center of the 1 km diameter antenna.

The factors affecting the temperature and temperature gradients in an orbiting structure are incident solar radiation and the heat generation within the structure. The influence of solar radiation is neglected in this analysis by assuming a "white" coating on the tubes (α_s 0.3, ϵ = 0.9). Because of the extensive infrared radiation area relative to the structural members and the high antenna temperature, the heat generated within the structure is the major factor affecting the temperature and the temperature gradients. The waste heat flux radiated from the antenna surface to the structure and the associated "effective" antenna radiation temperature are given in Figure IIC-15 as a function of distance from the center.

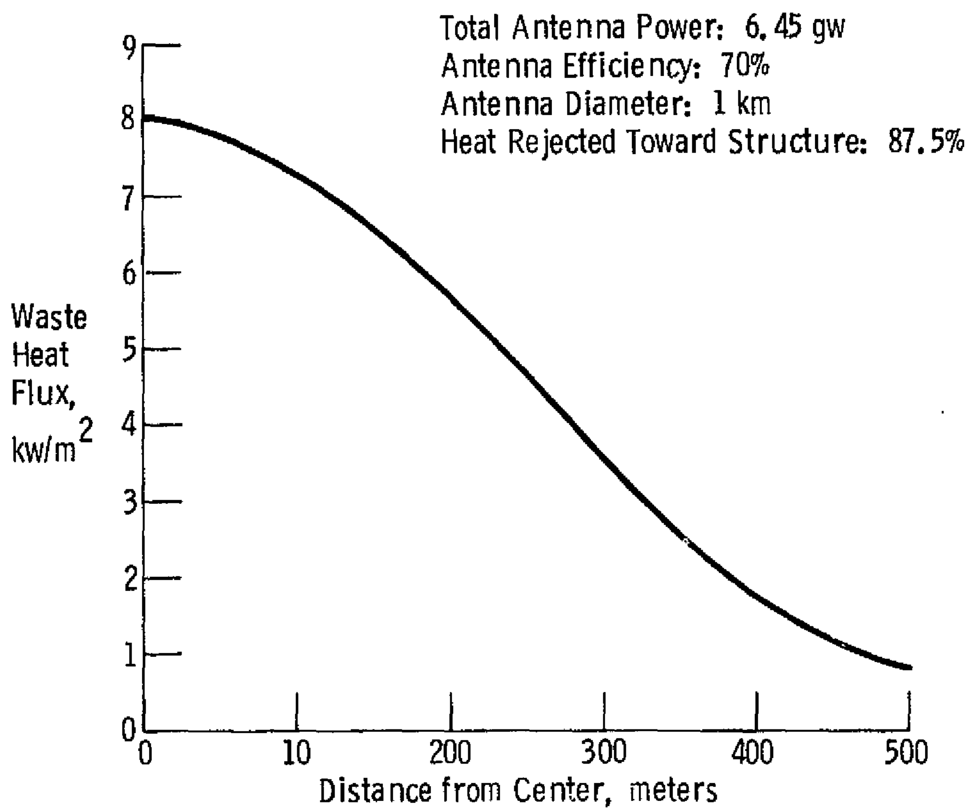


Figure IIC-15 Antenna Waste Heat Flux

From Figures IIC-15 and IIC-16, the effective antenna temperature near the center of the structure is in the vicinity of $600^{\circ}K$. Any truss member receiving radiation from this surface on one side and radiating to space on the other will attain an average steady state temperature given by $600/(2)^{1/4} = 500^{\circ}K$. Because the use of aluminum is inadequate for this application ($500^{\circ}K$), the structural members analyzed here are steel tubing having a wall thickness of 0.018 inches. Material selection is discussed after the thermal stress analysis below.

Columns - Because vertical tubular columns are equally radiated from all sides over their entire length, both the longitudinal and circumferential temperature gradients will be small ($<3^{\circ}K$). The actual temperatures will vary between the center and the edge of the antenna as shown in Figure IIC-17 because of the radial temperature gradient in the antenna surface itself.

Tubes - Circumferential temperature gradients within the horizontal tubular members were analyzed by the finite difference model shown below.

D = tube diameter
S = wall thickness
(A) refers to antenna

The steady-state energy balance equations for the model, assuming internal radiation is negligible relative to circumferential wall conduction, are

$$\sigma(\pi D/8)\ F_{na}\ \varepsilon n\ (T_n^{\ 4} - T_a^{\ 4}) + (8\ k\ \delta/\pi D)\quad (Tn\text{-}Tn\text{+}1) - (Tn\text{-}1\text{-}Tn)$$

$$+\ \sigma(\pi D/8)\ F_{ns}\,\varepsilon n\ (T_n^{\ 4}) = 0$$

where n = 1 through 8

S refers to space
σ = Stephen - Boltzmann constant

F1A = 1.00 F2A = F8A = 0.75 F3A = F7A = 0.50

F4A = F6A = 0.25 F5A = 0 F1S = $\frac{0}{1.00}$ F2S = F8S = $\frac{0.25}{1.00}$

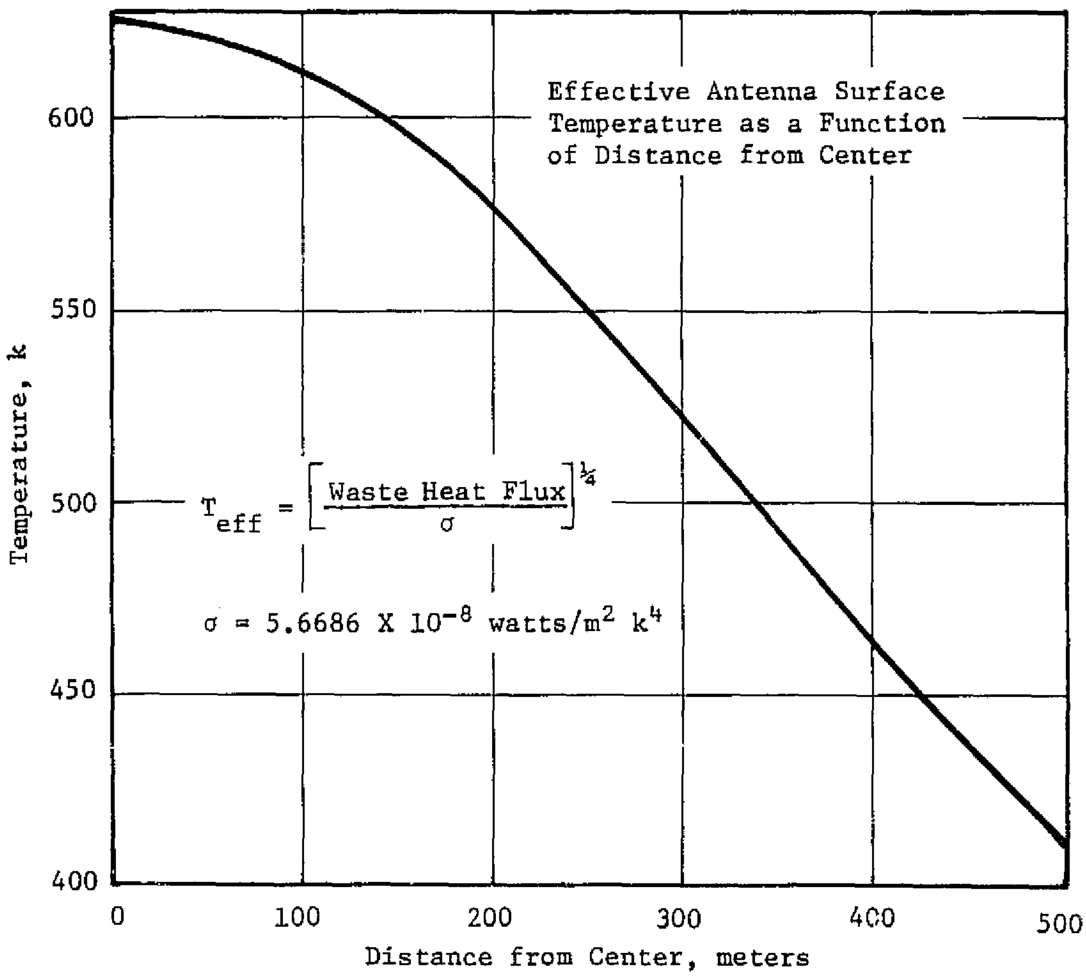


Figure IIC-16 Effective Antenna Surface Temperatures

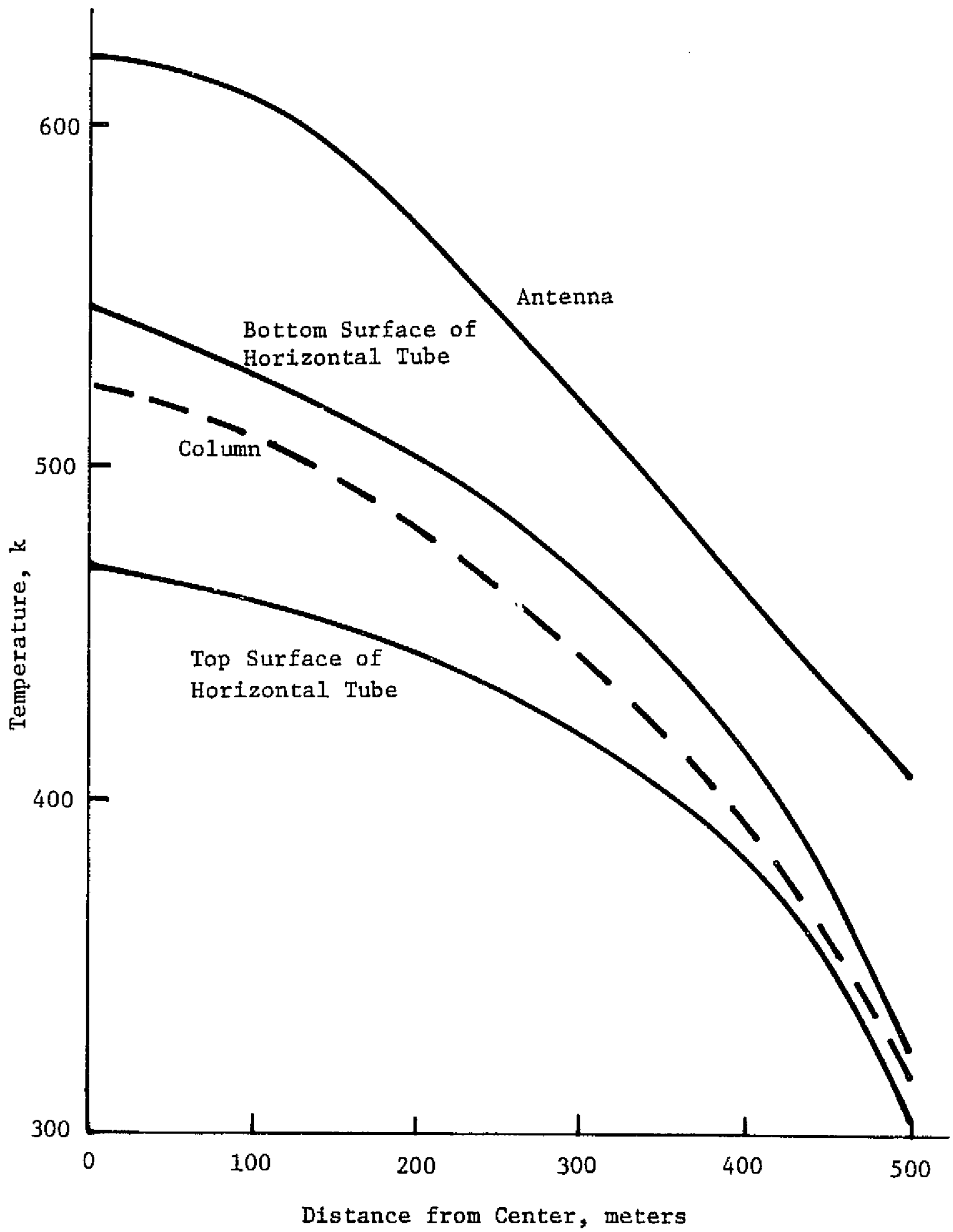


Figure IIC-17 Antenna Structure Temperatures

$F3S = F7S = \frac{0.50}{1.00}$ $F4S = F6S = \frac{0.75}{1.00}$ $F5S = \frac{1.00}{1.00}$

$\varepsilon_n = 0.90$

$D = 1\frac{1}{2}$ inches

$\delta = 0.018$ inches

T_a given by curve in Figure IIC-17.

This set of 8 equations is solved for the 8 Tn's. The resulting numerical values of T_1 (bottom surface) and T_5 (top surface) are plotted in Figure IIC-17 as a function of the distance from the center.

The set of energy balance equations can be written in the form

$$(Tn - Tn{+}1) = \left[\frac{\sigma(\pi D/8)^2 \varepsilon_n}{k\,\delta}\right]\left[F_{na}\,T_a^{\,4} - T_n^{\,4}\right] + (Tn{-}1 - Tn)$$

which indicates that the temperature gradient is a function of the square of the tube diameter.

The basic asumption that internal radiation is negligible relative to circumferential wall conduction is valid if

$$\frac{\sigma(\pi D/8)^2\, F_{12}\,(T_2^{\,2} - T_1^{\,2})\,(T_2 + T_1)}{k\,\delta} << 1$$

In this case, this ratio has the numerical value of

$$\frac{(0.1713)(1.5\ \ /96)^2\,(0.2)(6^2 + 5^2)(6+5)(1.8)^3\,10^{-2}}{26\ (0.018/12)} = 0.083$$

which indicates that internal radiation is small relative to circumferential wall conduction.

The temperature gradient between the upper and lower cap has particular significance in the determination of overall structural deformation. Due to the high heat flux in close proximity to the structure, this value is small, about 3^oK.

The consideration of the use of white coatings on structures that must have a long useful life should be evaluated. The ability of existing white coatings to survive the space environment for as long as 50 or more years is not understood.

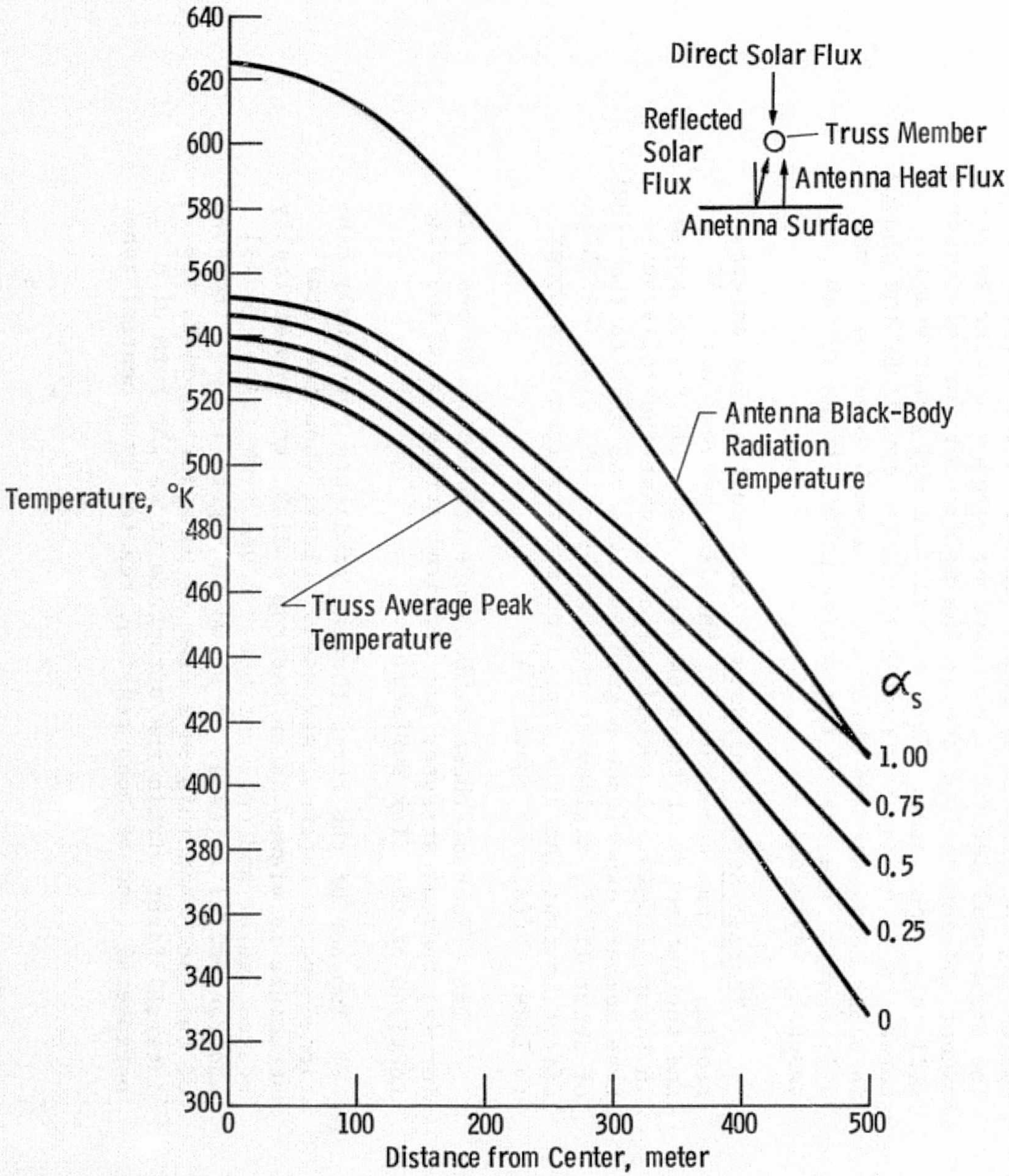
It is felt that other coatings more likely to remain effective over these long periods should be evaluated. If coatings of a higher absorbtivity are used, solar heating becomes more prominent as can be seen in Figure IIC-18. The increase in the average truss temperature at the center of the antenna is shown to be about 26°K, at an α_s of 1.00. In view of this small temperature increase it is felt that coatings other than white should be given strong consideration.

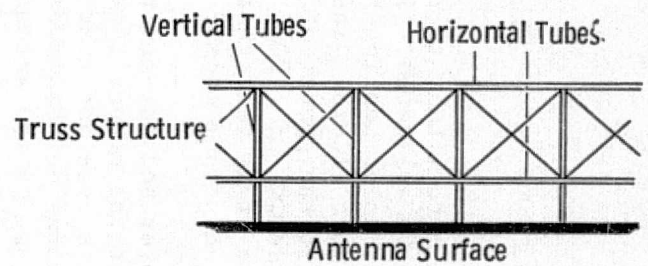
The two heating cases, solar and antenna radiation produce two conditions that affect the proper operation of the antenna. Antenna radiation is a constant heat source that once applied will cause structural deformations and stress that will remain constant as long as the antenna is being operated. Solar heating affects the structure on a 24 hour cyclical basis. Both heat sources must be considered when evaluating the structural loads imposed on the structure. The structural deformation caused by antenna heating is permanent and antenna surface aiming to correct for this error will only occur once. This occurs when the power station is turned on. The daily structural changes caused by the change in solar incidence and which are of a much lower magnitude will require the use of the active panel aiming system proposed by Raytheon.

Thermal Stress Analysis - The result of the antenna heating is that the center of the disk becomes hot (456°F) and the periphery of the disk stays cool (110°F), as shown in Figure IIC-19. A second effect is that the individual tube members experience a gradient from one side to the other of about 117°F, maximum at the center of the disk. A third effect is that the disk experiences a gradient from upper to lower surface of about 5.4°F.

The effect of these temperature gradients was evaluated for (a) thermal stresses and required strength and (b) distortions of the disk surface.

Two out of the three thermal gradients previously mentioned create internal stresses. First, when the total structure is viewed as an isotropic disk, the symmetrically heated center and cool periphery cause symmetrical radial and tangential stresses as shown in Figure IIC-19. Because of the fact that this is a free or unsupported disk, the internal fight within the structure takes the form of a peripheral band of hoop tension, reacted by a central zone





Structural Concept

Figure IIC-18 Tubular Truss Member Peak Average Temperatures

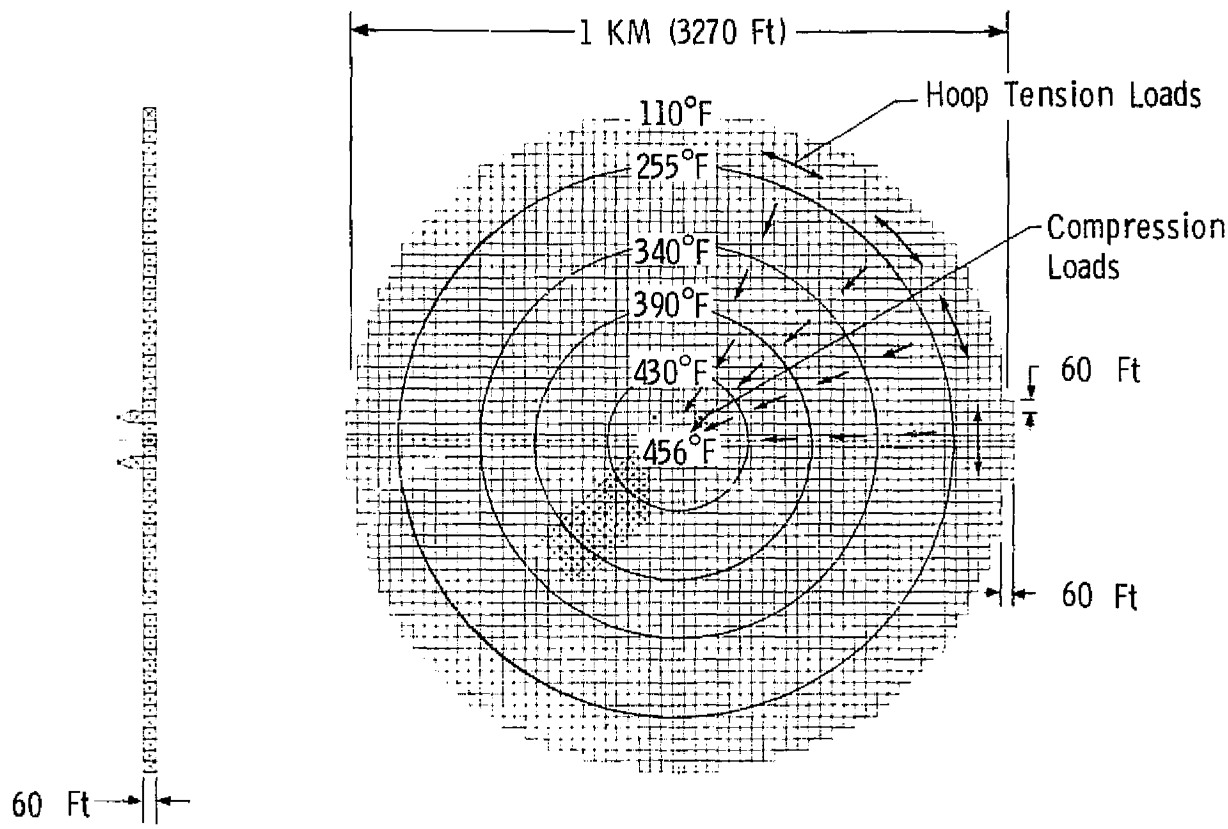


Figure IIC-19 Temperature Gradients

of radial and tangential compression. The second effect is the induced bending in individual tube members due to temperature differential across the tube itself. This effect is additive on one side of the tube to the "overall" stress created by the first effect. The third effect, temperature differential across the thickness of the total disk, creates only free distortions with no stress.

The stresses mentioned above were determined and are plotted in Figure IIC-20. The max stress shown for the tube is the additive effect of axial tube stress due to the "overall" disk loads and the local bending stress in the tube. It should be noted that the overall disk analysis assumes an isotropic circular plate. This assumption is believed to be sufficiently accurate for the purpose of this study.

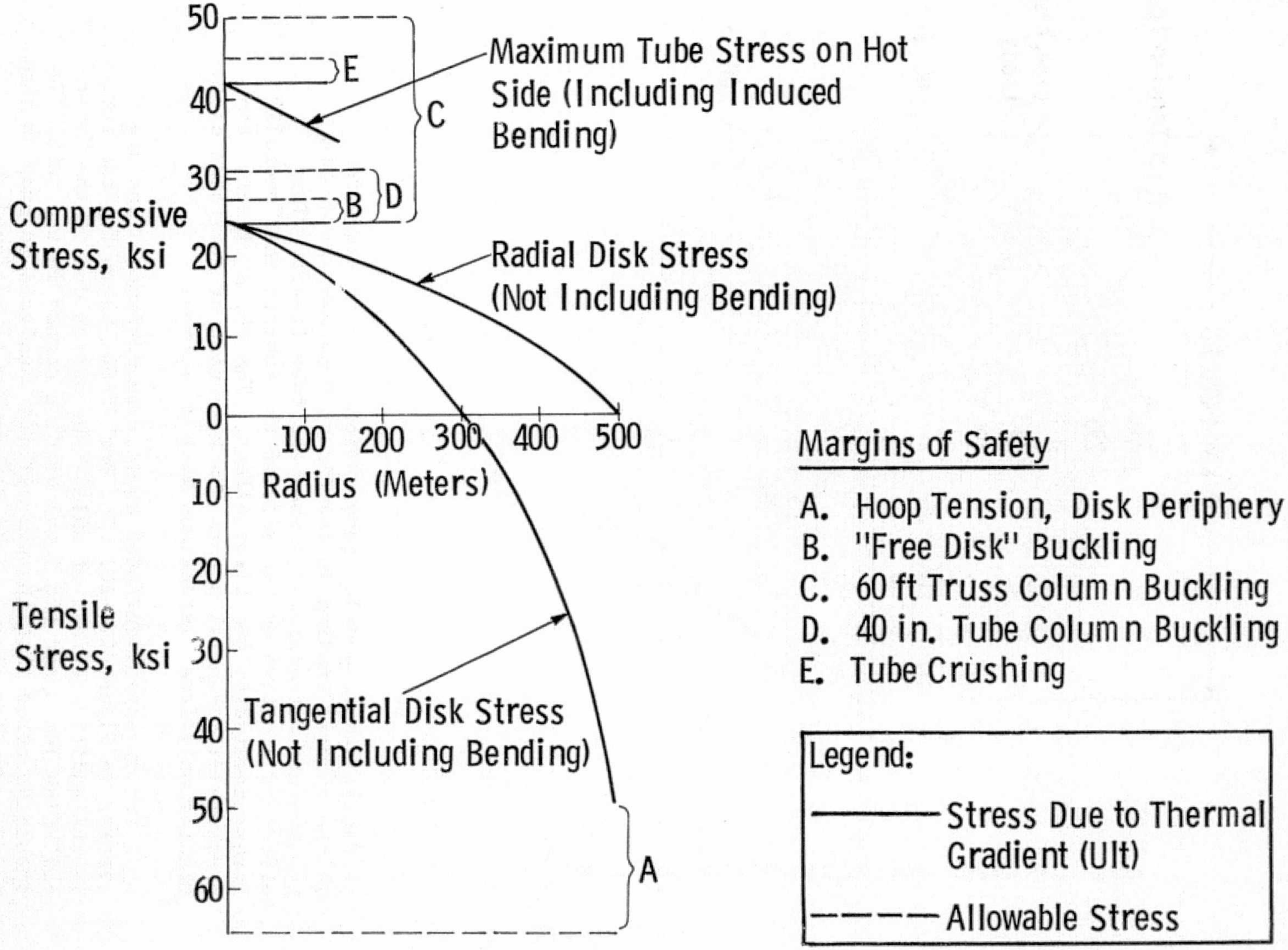


Figure IIC-20 Strength Summary

Figure IIC-20 also compares the computed stresses with allowable stresses. The allowable stresses shown are (a) compressive buckling allowables applicable in the high compression zone of the disk and (b) the tension allowable applicable for the hoop loads in the peripheral zone.

Allowables for all significant failure modes were computed. The compression allowables, ranging from the largest failure mode to the smallest, include the overall disk buckling, buckling of the 60 foot horizontal triangular columns which make up the disk, buckling of individual 40 inch tubular members which make up the 60-foot columns, and the local wall buckling, or crushing, of the 1-1/4 x .018 inch thin-wall tubes, which make up the horizontal tubular members. These modes of failure are shown pictorially in Figure IIC-21.

The tension allowable is simply the weld allowable of welded alloy steel. All allowables account for reduced material properties due to temperature.

Structural distortions are produced by the temperature gradients. The nature and magnitude of these distortions are shown in Figure IIC-22.

The 5.4°F temperature differential across the 60-foot thickness of the disk causes a bowing or dishing effect, which resembles a spherical segment of very large radius.

The hot center and cool periphery cause a shifting effect or shear displacement, and also a thickness change of the overall disk. The shifting effect is caused by the fact that the diagonal X-braces near the center of the disk should develop compression along with the main horizontal members. However, their compression capability is so low, they will buckle and a slight lateral shift of the bay will occur, until equilibrium is established. The accumulation of these shifted bays results in a total displacement of the center with respect to the periphery.

The effect of local moments in the tube members due to Δt across the tubes was checked and found to be insignificant, approximately 1/3 inch.

The maximum angular displacement is calculated to be 4.3 minutes at the outer edge.

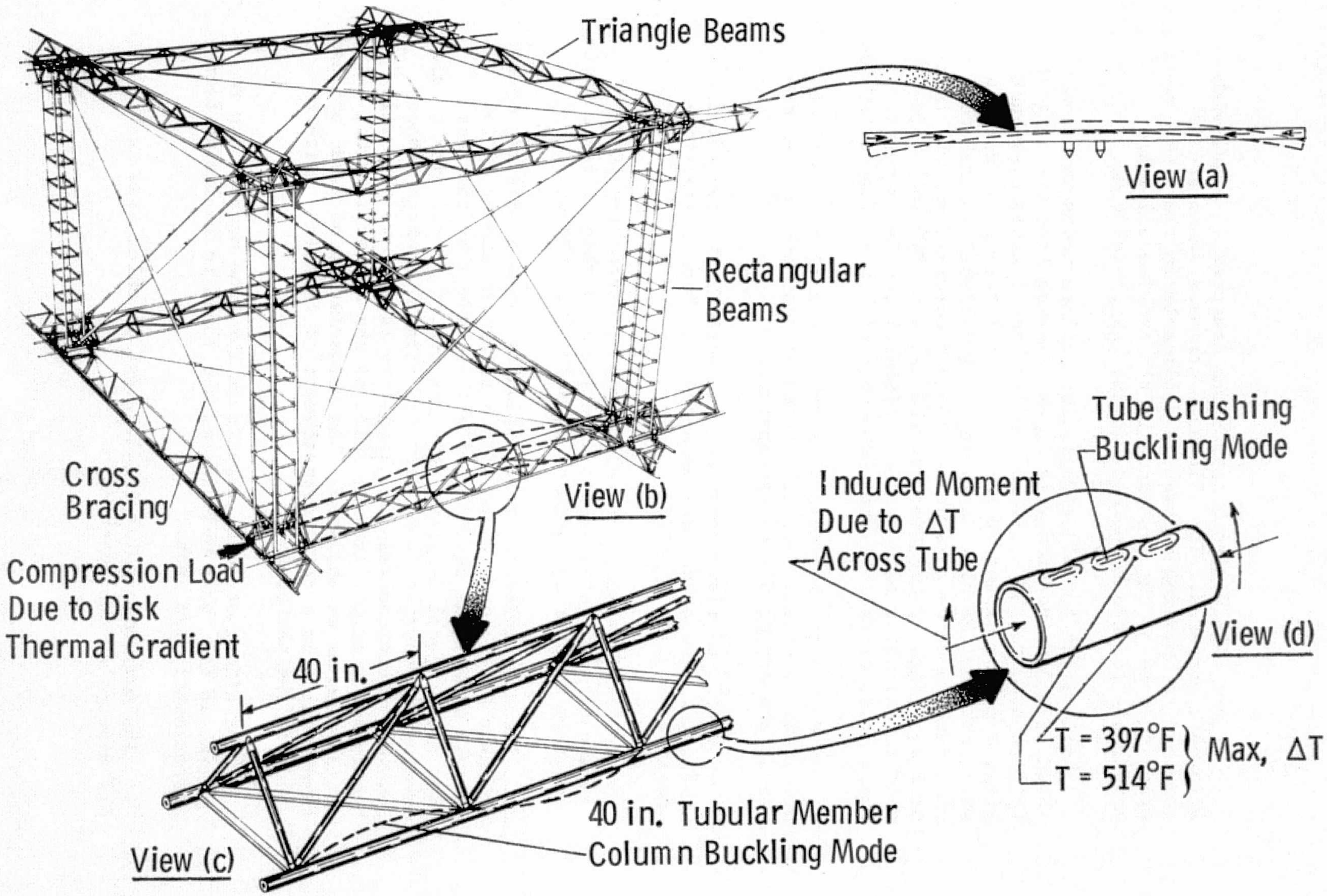


Figure IIC-21 Buckling Modes

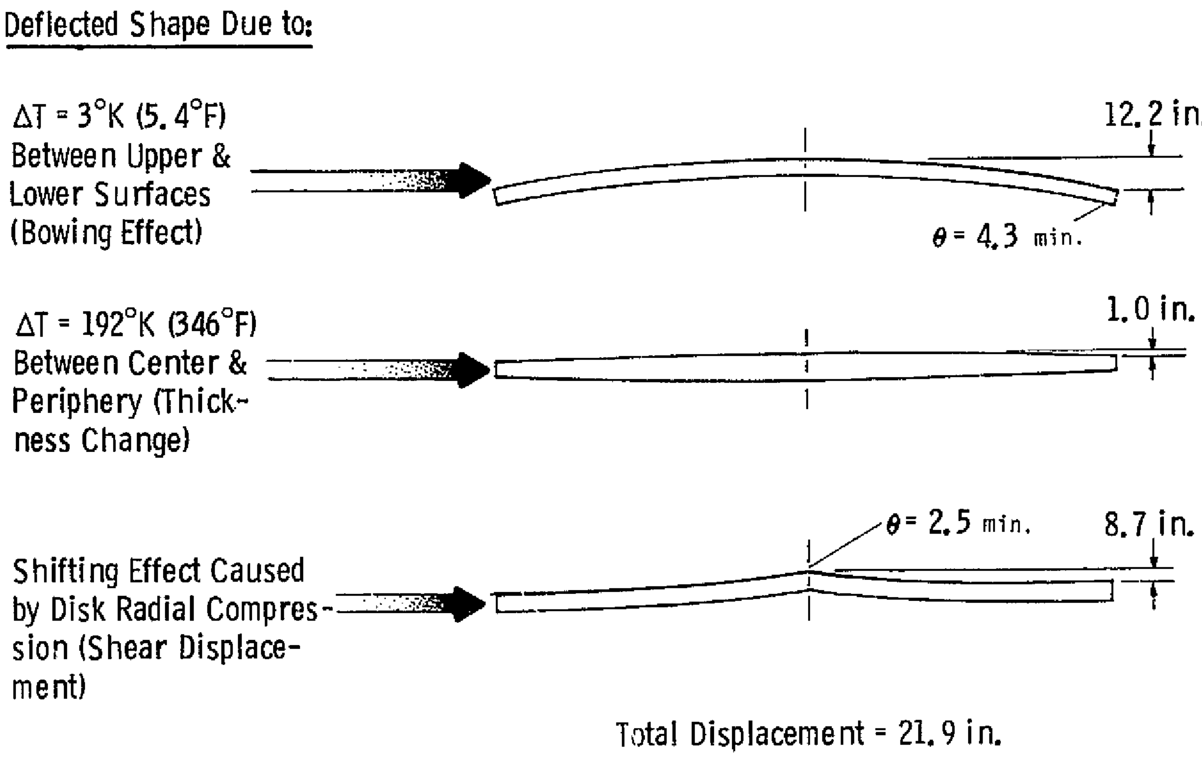


Figure IIC-22 Structural Distortions

In summary, the thermal/structural problems of a point design of an antenna support structure have been studied in sufficient depth to understand the problems and to gain confidence that the design is workable. A structure built of alloy steel tubing was analyzed for internal stresses and distortions due to thermal gradients. For the point design analyzed, positive strength margins exist at all points in the structure. Structural displacements due to temperatures are within reasonable bounds. Total out-of-plane deflection is calculated to be 21.9 inches and angular displacement is 4.3 minutes.

It should be emphasized that these results apply to a given design and thermal environment. If the design or environment were to change, it may be desirable to make configuration changes which would result in lower stresses and deflections. The stresses could be lowered to only a fraction of values in this study by selectively deleting

cross braces in the "corner" areas of the disk. In fact, due to the potential desirable results of brace deletions, further study is believed to be warranted on this subject.

Material Selection - The high temperature ($456^{o}F$) in the central area of the disk dictates the use of a material having good dimensional stability and high temperature strength. Aluminum, steel, beryllium, titanium, and non-metals, such as graphite polyimide, were considered as candidate materials as shown in Table IIC-3. Aluminum has inadequate strength and stability properties at the required temperatures. Beryllium and high temperature non-metals are expensive and not as versatile as aluminum, steel, and titanium from the standpoint of assembly and fabrication. Thermal stresses are high in beryllium, intermediate in steel and aluminum, and low in graphite composites or titanium. Thermal distortions are high for aluminum, intermediate for beryllium, steel and titanium, and low for graphite composites. Cost and ease of fabrication of alloy steel and aluminum are superior to all other candidates. Weight is not critical for this application due to volume limitations in the Shuttle cargo bay.

Table IIC-3 Materials Comparisons

Material	E 10^6 psi	$\alpha 10^{-6}$ in./in. °F	$E\alpha$	F_{cy}, ksi
Aluminum (2219)	8.6	13.5	116	18 5 (Weld)
Steel (Alloy)	26	7.5	195	61
Titanium	14	5.3	74	97
Beryllium	40	7.7	308	22
Graphite Poly	20	1.0	20	35

Properties Are at 500°F

Thermal Distortions Are a Function of α

Thermal Stress Is a Function of $E\alpha$

This thermal and stress analysis of the microwave antenna structural application points out the need for the proper selection of materials and coatings. The high operating temperatures of this application indicate that steel with some type of oxide coating would be a good choice for several reasons. They are:

(1) High strength at high temperature
(2) Low cost
(3) Ease of manufacturing
(4) Minimum weight penalty over other high temperature materials
(5) Manageable thermal expansion properties

Graphite/Polyimid has been suggested as a material that exhibits good strength and thermal expansion properties. MMC has done a substantial amount of work with this composite material over the last several years and has gained much experience in the proper use, fabrication and assembly techniques. With this understanding, we believe that a more thorough evaluation of this material for the microwave antenna application is indicated as an alternate to steel.

For applications other than the microwave antenna, aluminum would be considered providing high temperatures were not involved and a proper long life thermal coating is available.

2. Packaging for Delivery to Orbit

a. Structural Members - The length of the cube design discussed above was strongly influenced by the Shuttle Orbiter cargo bay dimensions. A ground rule of this study specifies Shuttle as the transportation vehicle since future launch vehicle concepts are not well defined at this time.

In order to maximize the available Shuttle payload, some form of efficient packaging of structural members during transport is required. Three basic approaches were studied.

The first approach looked at was to transport totally fabricated rigid beams in the Shuttle cargo bay. While this approach assures the structural integrity of each member and reduces the fabrication cost of the structural members, it appears to be unsuited for the task. The geometry of the structural members creates an extremely volume-limited payload condition, which would not be cost effective in terms of the number of Shuttle flights required to transport the members to orbit.

The second approach to the problem is to use collapsible members for fabrication of the antenna support structure. While in transit, the structural members are folded down on themselves as depicted in Figures IIC-23 and IIC-24. This is accomplished by means of pivots on two sides of each beam, allowing the upper side (in the case of the square beam) or upper corner (in the case of the triangular beam) to lie flat against the lower side.

In order to permit collapsing the beams in this manner, the cross braces of two sides of each beam must be capable of varying in length. This is accomplished by using telescoping members for the braces (Figure IIC-25 top). Once in the deployed position, pyrotechnic locking devices, which are discussed later, lock the telescoping braces into rigid members. This fixes the beam in its deployed configuration.

Also, in this approach, the cross braces required for the fabrication of the antenna support structure are of a telescoping design, with pyrotechnic devices employed to lock the brace after final alignment of the beam. The stowed cross braces (Figure IIC-26), 57 feet in length, are attached to a beam on one end prior to assembly of that beam into the structure.

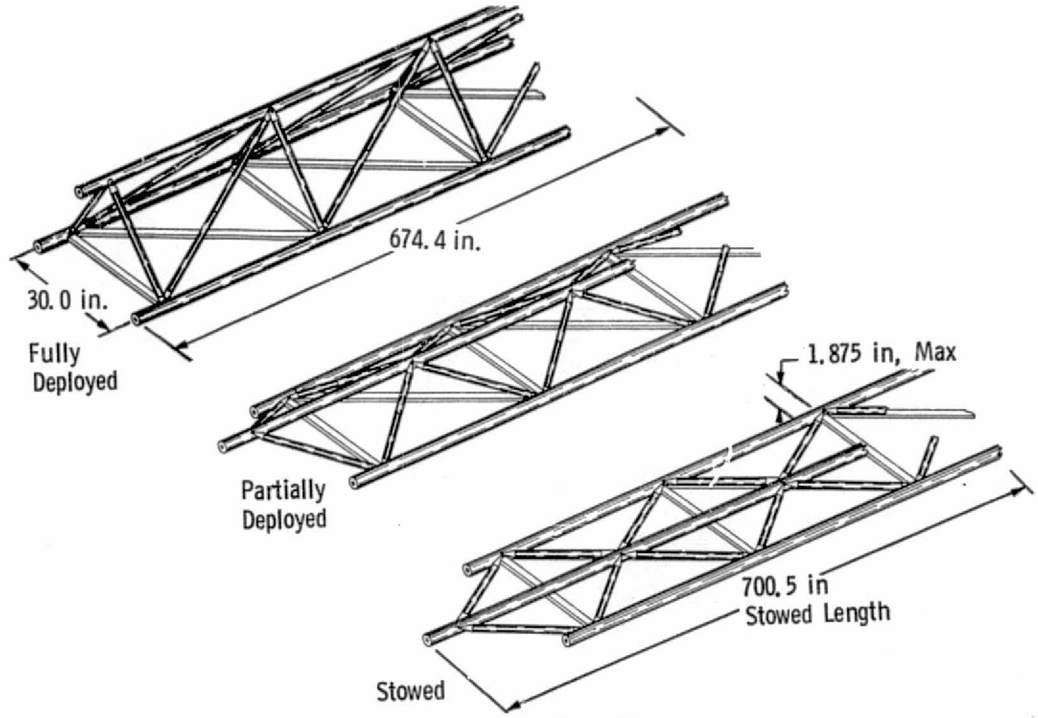


Figure IIC-23 Collapsible Triangular Beam

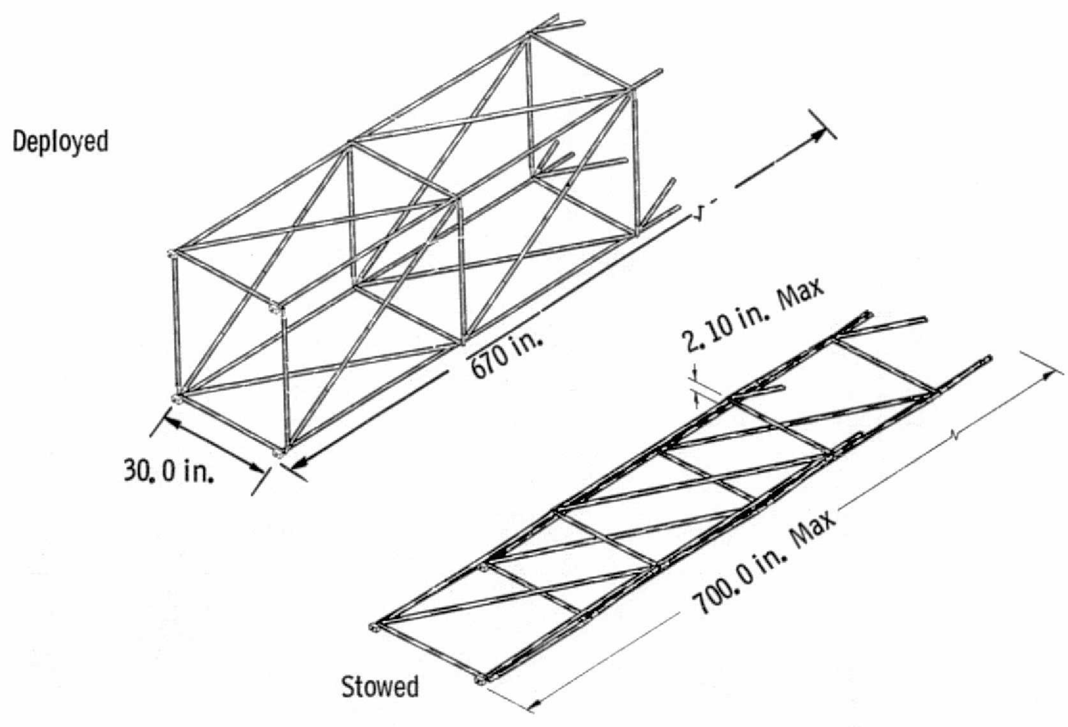


Figure IIC-24 Collapsible Square Beam

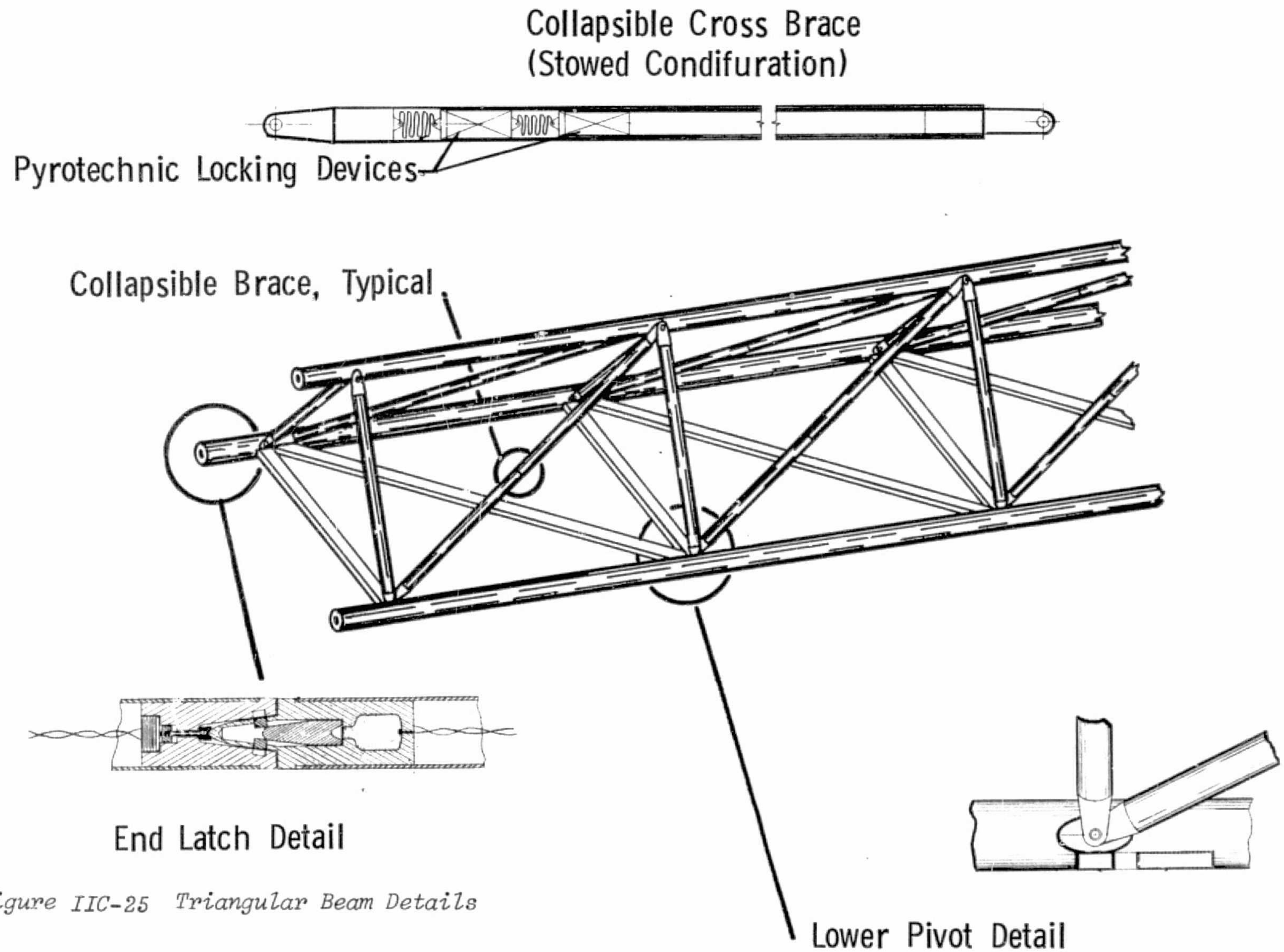


Figure IIC-25 Triangular Beam Details

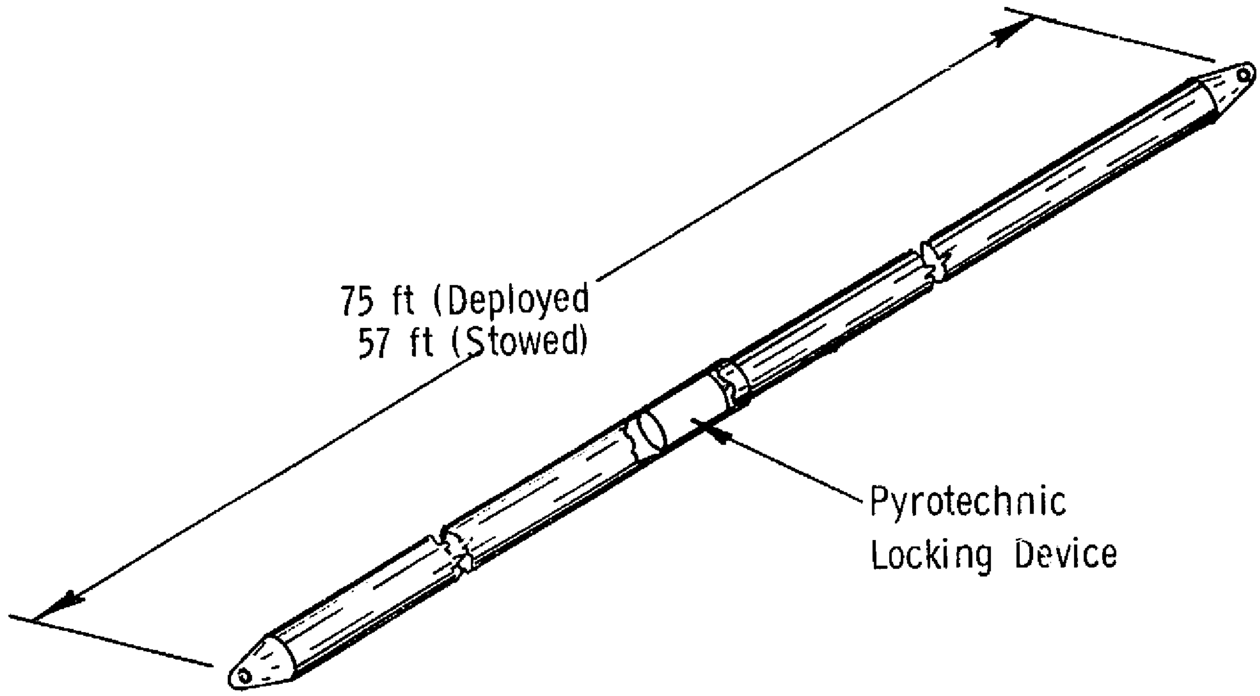


Figure IIC-26 Telescoping "X" Brace

Once the beam has been positioned, the brace is extended and the other end attached to the structure. The beam is then aligned into its final position and once verified, the pyrotechnic device in the cross brace is activated, locking both the brace and the beam in position.

With the introduction of telescoping members into the structural design, it became necessary to utilize a device for locking those members once in the proper configuration. This is accomplished by means of pyrotechnic pin throwers located in the telescoping members (Figure IIC-27). This device, when activated, pushes two pins through holes in the wall of the inner tube and through the wall of the outer member. The device is activated only when proper alignment of the members involved has been verified.

The pin throwing technique has advantages over many others, as it allows infinite adjustment of the member prior to locking; i.e., there are no incremental restrictions placed on length, as would be the case if holes in each telescoping member had to be lined up prior to locking.

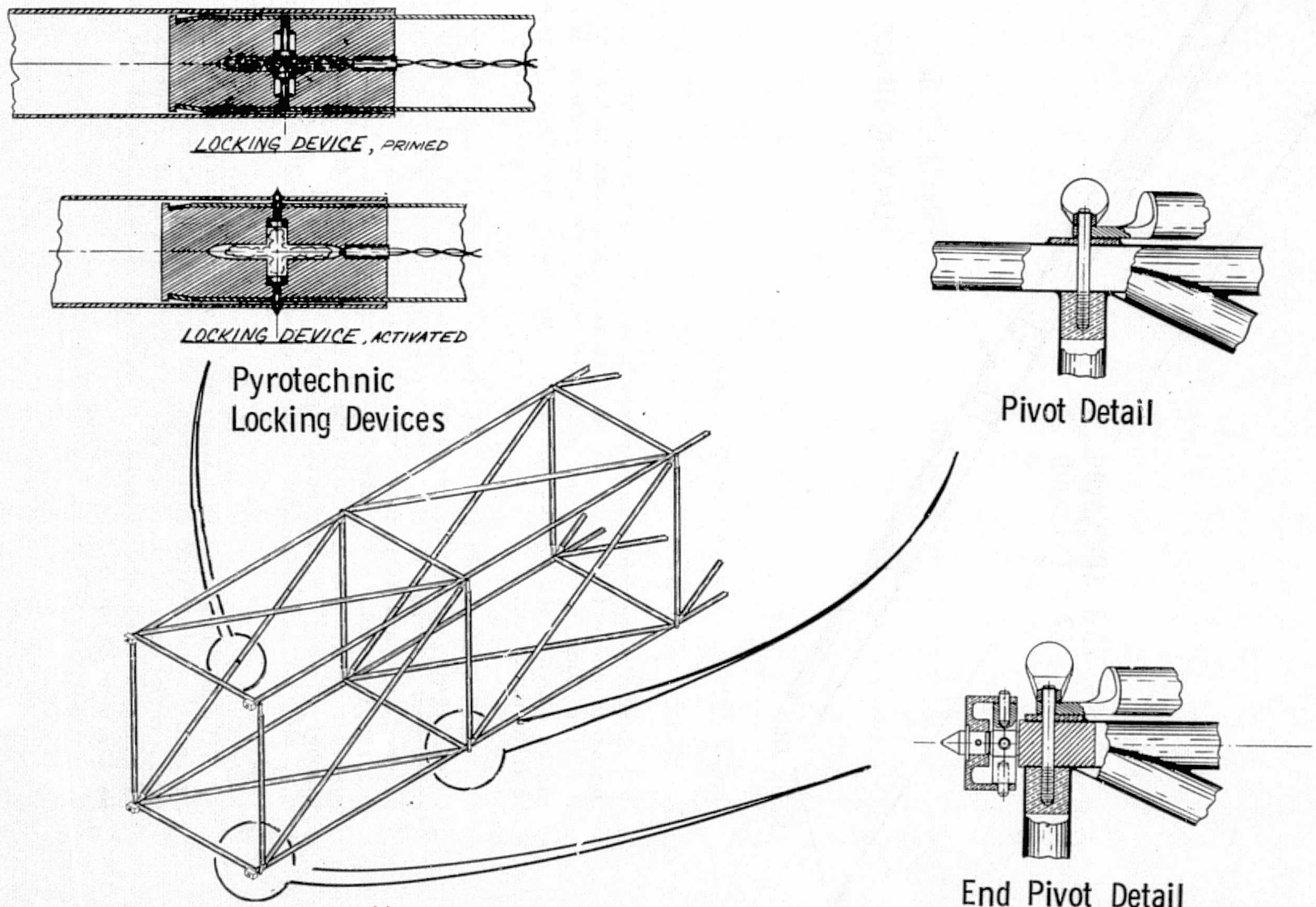


Figure IIC-27 Square Beam Details

b. Beam Packaging and Dispensing Pallet - Due to the packaging and dispensing requirements of the structural members with this approach, a specialized pallet was developed to serve as a storage package on the ground, in transit (Shuttle), and in orbit as a beam dispensing unit for the mobile assembler.

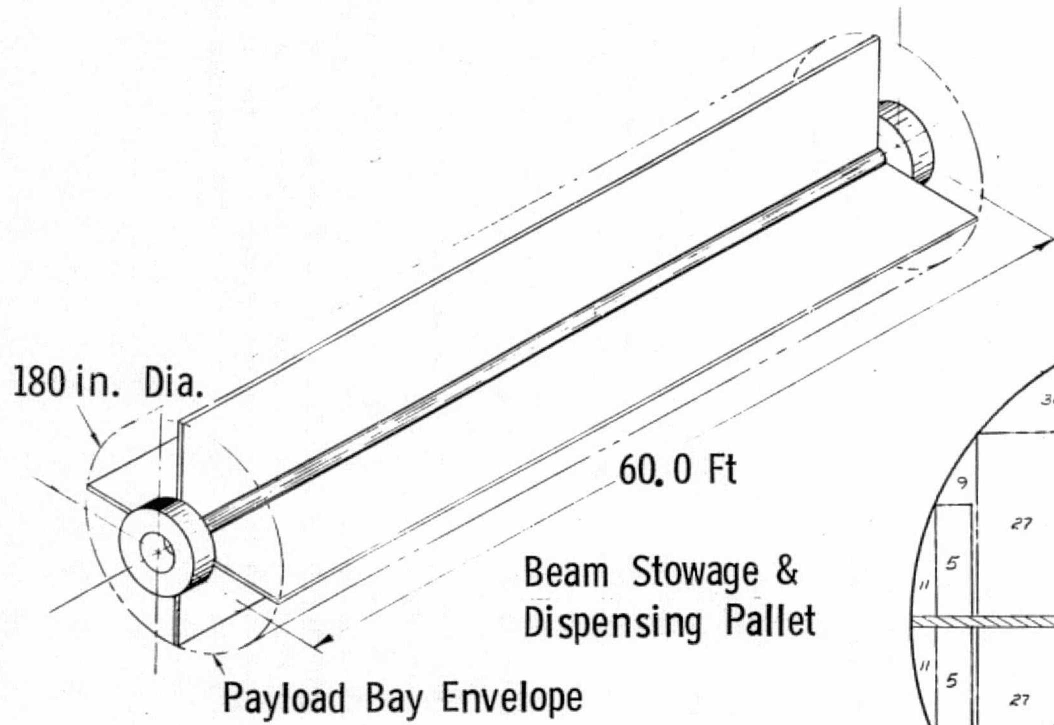
The pallet (Figure IIC-28) basically consists of a central tube with docking rings at each end. Four structural dividers extend radially outward from the central support tube to a diameter of 180 inches.

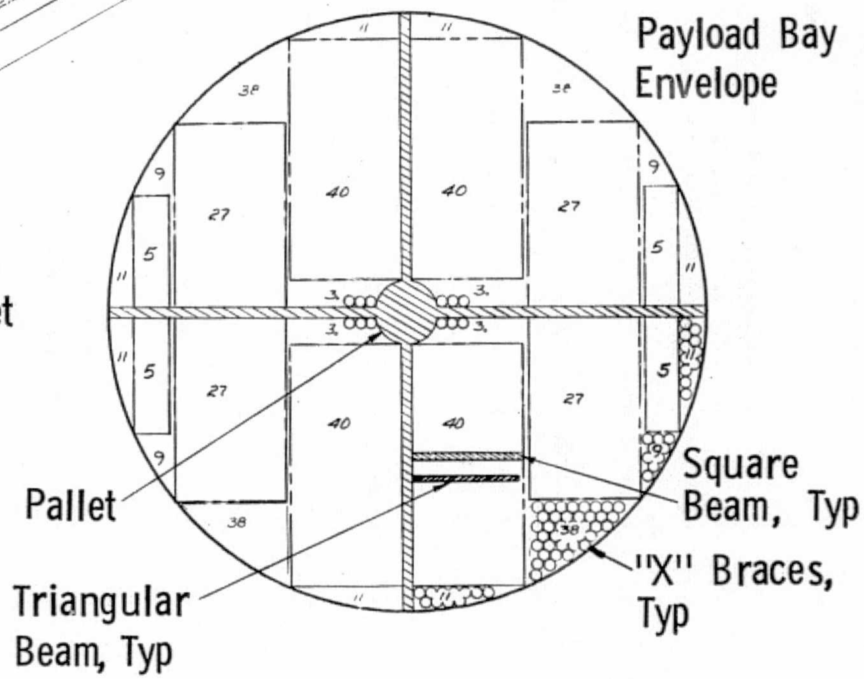
Collapsed beams and cross members are stowed in each of the four quadrants as shown in Figure IIC-28. Proper mixing of the structural members is predetermined and the quadrants are packed so that the member needed by the mobile assembler is available in the proper sequence. Each pallet (or Shuttle payload) contains 192 triangular beams, 96 square beams, and 288 cross braces. This results in a total structural member weight of 35,870 pounds. Adding an estimated 2000 pounds for the weight of the pallet, the total Shuttle payload becomes 37,870 lbs.

With this system of packaging, a payload density of 3.59 lb/ft^3 is attained. Shuttle has a payload density capability of 6.15. Therefore, this approach is still volume limited.

The beam packaging and dispensing pallet follows behind the mobile assembler base. A docking port is located at each end of the base, permitting docking of a full pallet without necessitating removal of the empty pallet until the Tug is free of the new pallet. Each docking port rotates to bring the proper pallet quadrant into position to permit access by the assembler.

A possible supplemental transport method for the structural members may be to transport a portion of them in the external Shuttle tank. With 35,000 lbs of beams in the caro bay, an additional 30,000 lbs of beams would be stowed in the external tank, thereby forming a full payload for Shuttle. In the present configuration, however, the structural members would reduce the fuel capacity of the external tank by approximately 25%, and the presence of pyrotechnic devices in a fuel tank would be undesirable. It seems feasible to assume, however, that with some modification of the structural members, this approach could prove to be effective from a cost savings point of view.





Payload Bay Cross Section, Beam Package
- 96 Square Beams 192 Triangular Beams, 288 "X" Braces
- Beam Weight - 35,870 Lbs
- Pallet Weight - 2,000 Lbs

Figure IIC-28 Beam Pallet and Shuttle Payloads Section

The third approach looked at involves the fabrication of structural members in an orbiting "factory" (probably manned) from raw material transported to the factory by Shuttle. This approach makes full use of the Shuttle payload capability (i.e., payload capability is no longer volume limited) and eliminates the need for rendering the beams collapsible. A cost study must be made to determine whether this approach is more cost effective than the collapsible beam approach.

3. Alignment Concept

Accuracy of alignment of the central core is assured by two methods: (1) ground test erection and alignment with precision tools, and (2) verification of alignment (and necessary adjustments) when assembled at the orbiter, through optical sightings by EVA crewmen.

Alignment of the outlying cubes will be achieved by adjustments, based on optical sightings, as the beams are fastened. The assembly will proceed in a spiraling manner such that two types of cube-assemblies will occur. Referring to Figure IIC-29, cube 1 (full cube) will require assembling 3 sides. Cubes such as 2 and 3 (partial cubes) require assembling 2 sides.

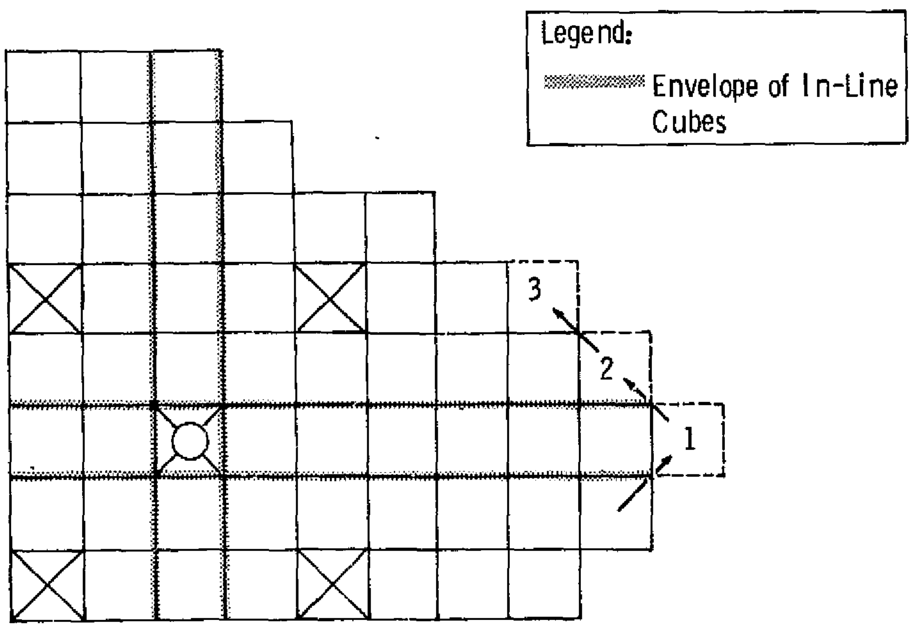


Figure IIC-29 Structure Assembly Progression

The cubes directly in line with the core section will be assembled very accurately. This is achieved by use of sighting points (bench marks) installed on the core section. Video cameras are installed on each end of the assembler bases (see Figure IIC-30). These cameras will have 5 degrees of freedom ($\pm$ X, $\pm$ Y, $\pm$ Z, pitch, yaw). When the assembler is at a location in line with the core, the cameras will be adjusted to perfectly align with the core bench marks. The cameras can then be pointed to desired points-in-space for defining required beam placement during assembly (see Figure IIC-31). These sightings will assure that installed beams are aligned with the core section. No alignment procedure prevents distance discrepancies due to beam length tolerance buildup. A tolerance of +0.030 in. at each beam would result in +1 in. error at the rim. However, tolerances would be random and the net error should be slight, and acceptable.

The partial cubes may be less accurately aligned, with greater reliance on the accurate installation of the full cubes and on manufacturing accuracies. The partial cubes will be leveled relative to the central core. This is achieved by first leveling the cameras with the core bench marks (see Figure IIC-32). The beams will be aligned by cross-sightings of both assembler end cameras. Final cross-sightings will be taken of the assembled cube. Errors will be stored in a computer base and used to correct alignment sightings when that beam is later used as the assembler base location.

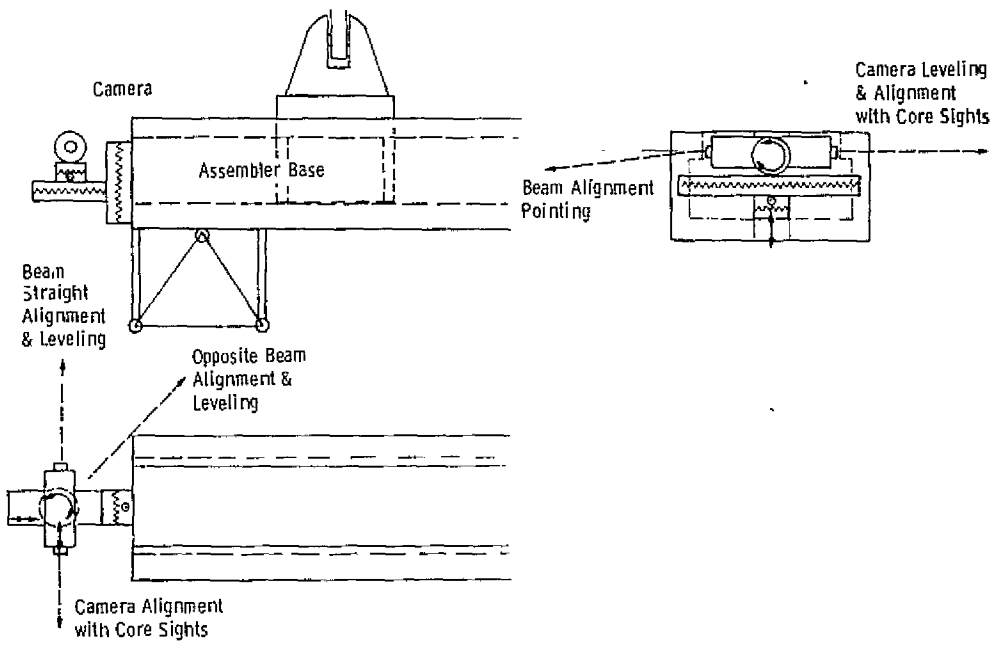


Figure IIC-30 Assembler Camera Installation

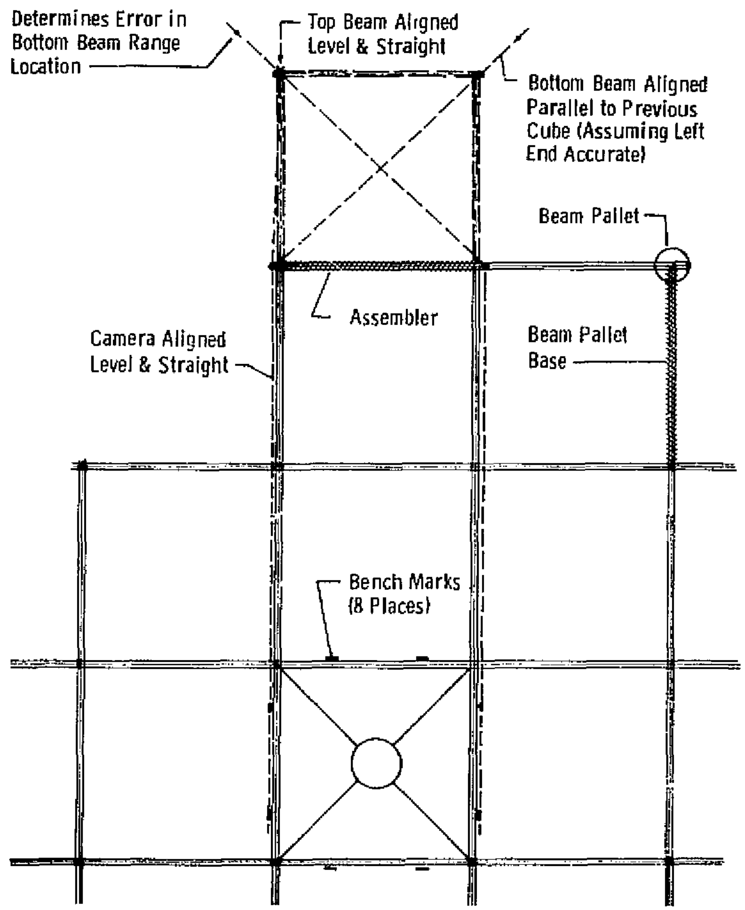


Figure IIC-31 Full-Cube Alignment

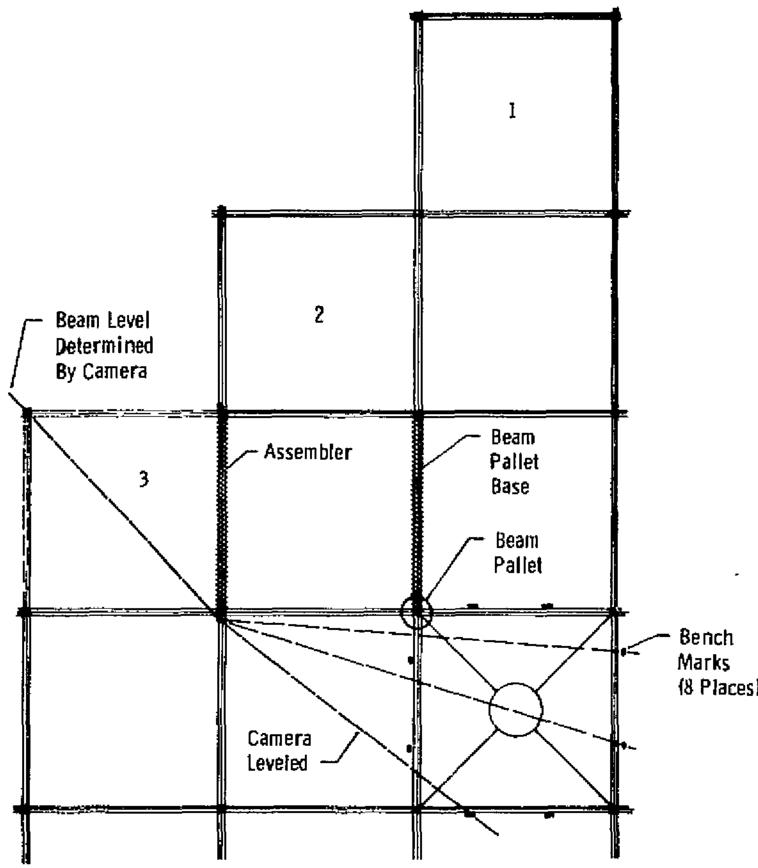


Figure IIC-32 Partial-Cube Alignment

4. Assembly Support Subsystems

Requirements for subsystems to support the assembly operations were investigated to the extent necessary to determine system feasibility and any problem areas.

a. Communications - Uplink commands are transmitted from the remote ground stations, either in the NASA Satellite Tracking Data Network (STDN) or the Air Force Space-Ground Link Subsystem (SGLS) network. Selection of the respective ground station network will determine the type transponder to be used in the core instrumentation. The STDN system provides communication interface capability with the Shuttle Orbiter and the Tracking Data Relay Satellite System (TDRSS). Although both the STDN and the SGLS utilize pseudo random noise (PRN) coding for identification and acquisition, transponder application is restricted to only one system due to the modulation format. The STDN uses staggered quadriphase PSK while SGLS uses bi-phase PCM. Selection of the STDN provides the use of standard NASA components such as the transponder and the AOP computer. In addition, communications support is available from two relay satellites in synchronous orbit to provide uplink commands or down-link telemetry data.

Referring to Figures IIC-33 and IIC-34, commands are received by the omni antennas simultaneously by being arrayed through the power combiner. The antennas are broad-beam, circularly polarized with an on-axis gain of +4 db and a HPBW of 120 degrees. The uplink commands have a discrete PRN code that is verified by the transponder and assures link acquisition through the down-link telemetry. On acquisition, commands are transmitted in real time to the on-board computer for storage or distribution via the command data bus.

The core antennas are deployed to extend beyond the faces of the structure. Dual systems are provided to prevent structure RF interference. The antenna boom mechanisms incorporate capacitive coupled rotary joints.

Commands for the assembler are transmitted through a secondary core transmitter and helix array subsystem to the antenna arrays, diametrically stationed such that commands are available to either assembler. Since the maximum transmission distance from the core to the assembler will be approximately 1/2 kilometer, 2 watts of RF power (or less) will close the link.

Reception of commands is provided by the command control subsystem provided for each assembler (Figures IIC-35 and IIC-36). Each subsystem will be activated by its assigned discrete address code. A total of 288 commands are available for assembly functions

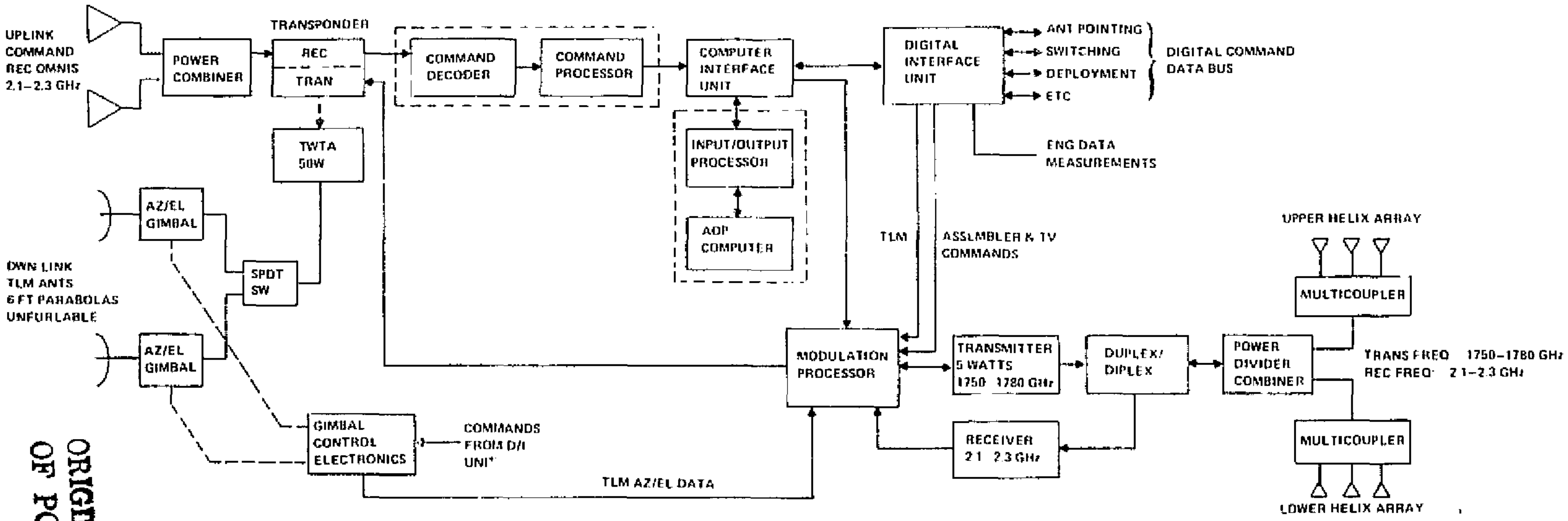


Figure IIC-33 Core Instrumentation

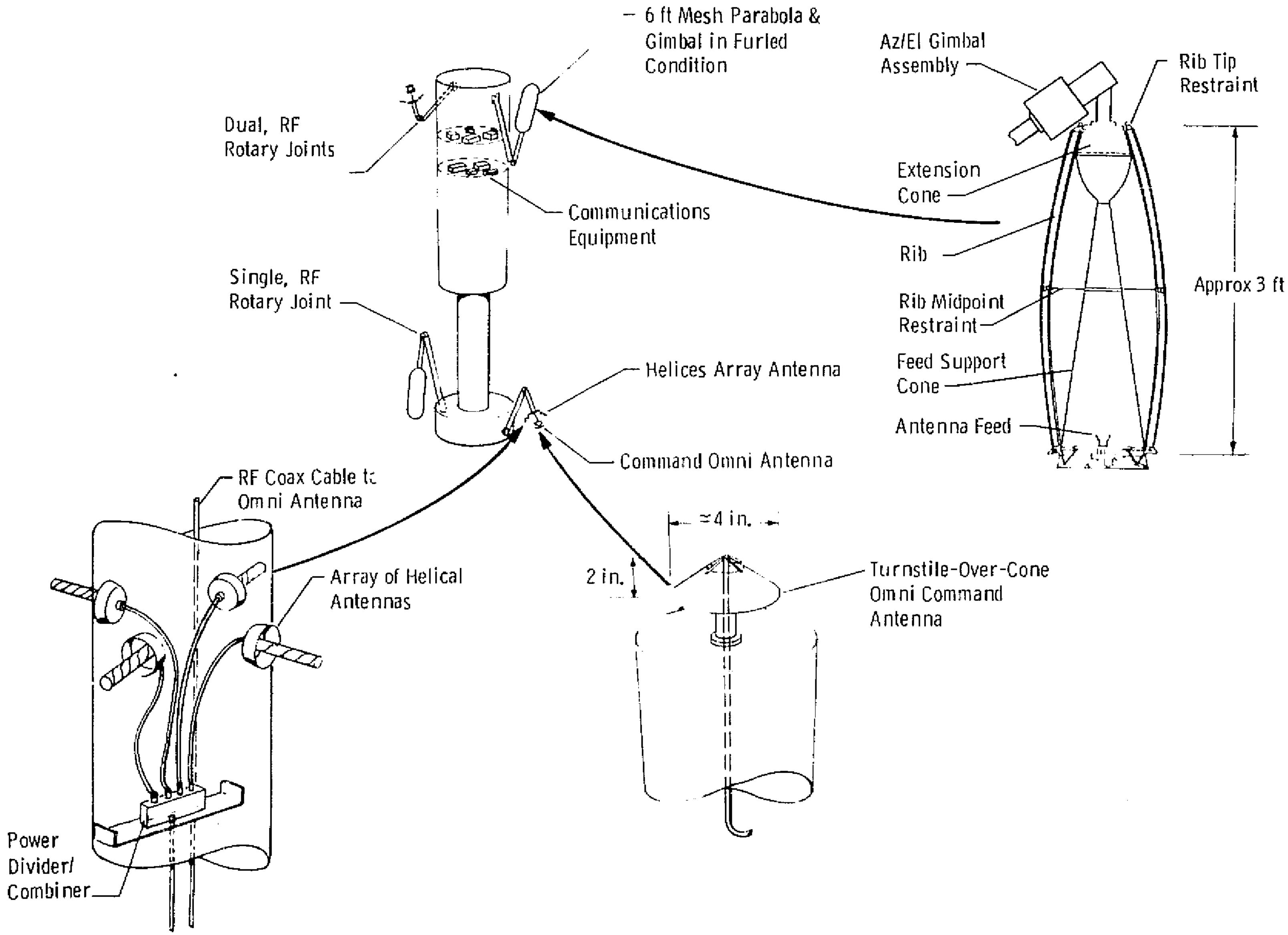


Figure IIC-34 Core Assembly Antenna Installation Concept

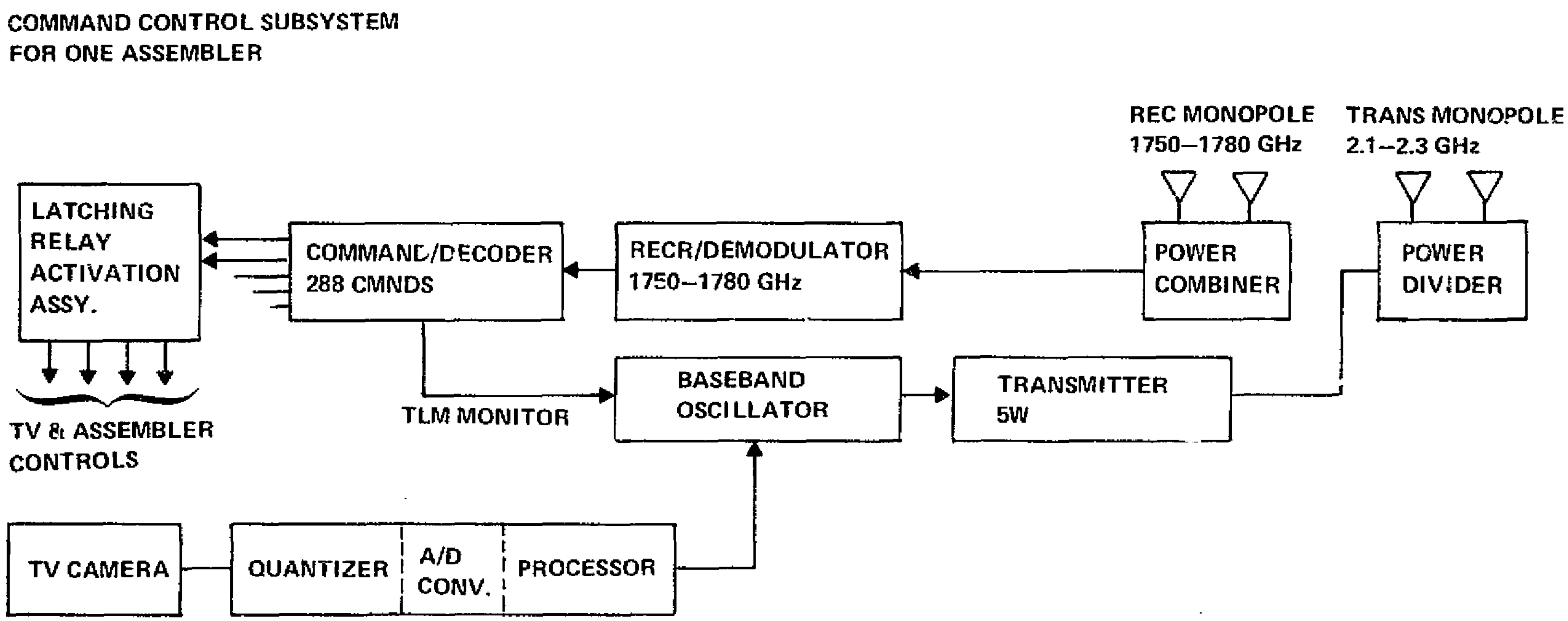


Figure IIC-35 Assembler Instrumentation

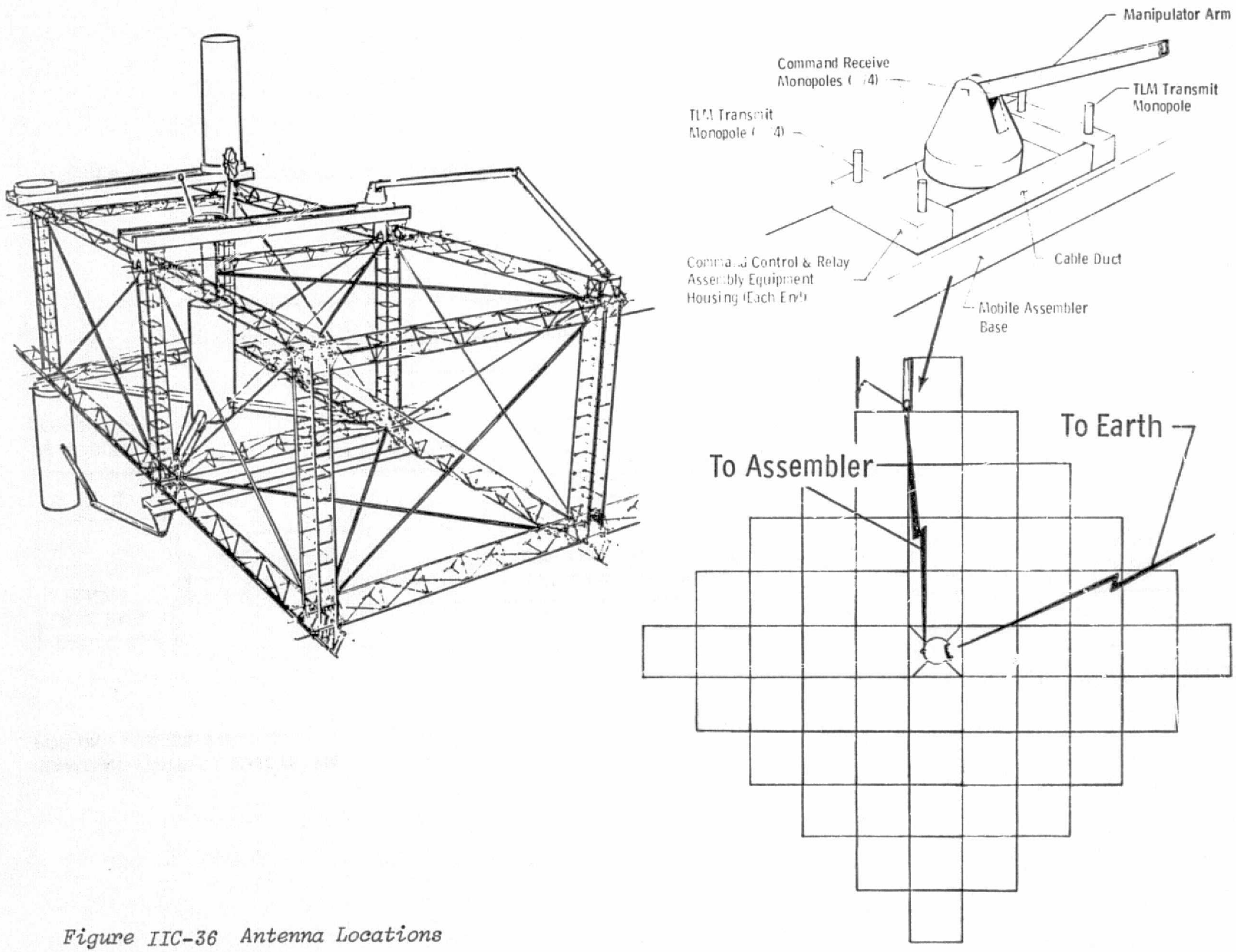


Figure IIC-36 Antenna Locations

which are activated by relay logic. Monitoring of the command functions is available from the TV cameras and the down-link telemetry which is remotely transmitted to the core communications system for processing and retransmission to the ground station with other housekeeping data. The down-link antennas are unfurlable, parabolic antennas, 4 to 6 feet in diameter (Figures IIC-34 and IIC-36).

Table IIC-4 presents the equipment list for the core and assembler instrumentation.

Areas Needing Further Definition

1. Data rate requirements for commands and down-link engineering data;

2. Communications interface requirements with Shuttle Orbiter, STDN, SGLS, TDRSS;

3. Storage capability of on-board computer based on the degree of autonomy required;

4. Degree of redundancy in the communications system to provide backup functions;

5. Extent of instrumentation and onboard monitoring of housekeeping functions;

6. Amount of TV resolution since this has impact on TLM bit rate and bandwidth;

7. Perform preliminary link calculation to establish baseline for data rates, antenna gain, and RF power requirements.

b. Electrical Power Systems - Electrical power must be provided at the core for operating subsystems; at the beam carriages for carriage translation, beam pallet rotation, and subsystems; and at the assemblers for carriage translation, manipulator operation, and subsystems operation. Estimated power requirements are presented in Table IIC-5.

Table IIC-4 Communications Equipment List

COMPONENT	DIMENSION	WEIGHT	POWER
CORE INSTRUMENTATION			
1. Receive omnis, turnstile over cone antennas	≈ 3.5 in. diam. x 3.0 in.	6 oz	Passive
2. Power combiner	2½ in. x 1½ in. x 0.5 in.	3 oz	Passive
3. Transponder: NASA/MSFC Std. Single Access or Multiple Access Options for Acquisition	7 in. x 4.5 in. x 2.75 in.	6 lbs	6 w @ 28vdc for RF output of 21mv
4. Command Decoder/Processor	TBD	TBD	
5. Computer Interface Unit	TBD	TBD	
6. Input/Output Processor	5 in. x 3.5 in. x 1.6 in.	≈ 2 lbs	(≈ 4 w)
7. Advance Onboard Computer (AOP) 8K work capability, plated wire memory. NASA Std. Component.	10 in. x 6 in. x 2.5 in.	≈ 5 lbs	(≈ 9 w)
8. Digital Interface Unit	6 in. x 4 in. x 3 in.	2.5 lbs	≈ 4 w
9. Modulation Processor	TBD	TBD	TBD
10. Transmitter, 5 w RF 1750-1780 GHz	5.7 in. x 4.7 in. x 1.4 in.	32 oz	2.5 amps @ 28 vdc
11. Duplex/Diplex	5 in. x 3 in. x 1.5 in.	≈ 10 oz	Passive
12. Receiver	5.5 in. x 3.6 in. x 6 in.	5 lbs	1.5 w
13. Power Divider/Combiner	2½ in. x 1½ in. x 0.5 in.	3 oz	Passive
14. Helix Antenna Array (upper and lower) with multicoupler	Array dimensions TBD. Each helix ≈ 3 to 4 in. long and 1/2 in. diam.	≈ 2 lbs per array and multi-coupler	Passive
15. Traveling-Wave-Tube Amplifier (50 w)	6 in. x 15 in. x 4 in.	9.5 lbs	145 w
ASSEMBLER INSTRUMENTATION (BEAM CARRIAGE SIMILAR)			
1. Receive Antenna Monopoles	$\lambda/4$; ≈ 1.5"L x ¼"D	≈ 3 oz	Passive
2. Transmit Antenna Monopoles	$\lambda/4$; ≈ 1.2"L x ½"D	≈ 3 oz	Passive
3. Power Combiner	2½ in. x 1½ in. x 0.5 in.	≈ 3 oz	Passive
4. Power Divider	2½ in. x 1½ in. x 0.5 in.	≈ 3 oz	Passive
5. Command Receiver Demodulator	5.5 in. x 3.6 in. x 6 in.	5 lbs	1.5 w
6. Command Decoder	5.2 in. x 7 in. x 3 in.	4 lbs	6.8 w
7. Relay Assembly	TBD (MMC)	TBD	TBD
8. TLM Baseband Oscillator	(≈ 5 x 4 x 1½)	(25 lbs)	(3.75 w)
9. TV Camera	TBD	TBD	TBD
10. Quantizer, A/D Converter, Processor	6 in. x 4 in. x 4 in.	≈ 3 lbs	≈ 2 w
11. TLM Transmitter (5 w)	5.7 in. x 4.7 in. x 1.4 in.	32 oz	2.5 amps @ 28 vdc

Table IIC-5 Assembly Support Power Requirements

		Power (watts)
<u>Assembler (each)</u>		<u>3780</u>
Manipulator and carriage	3000	
Subsystems	250	
Power conditioning and line losses	100	
Battery charging	250	
Solar array degradation allowance	180	
<u>Beam Carriage (each)</u>		<u>970</u>
Carriage translation and pallet rotation	750	
Subsystems	75	
Power conditioning and line losses	5	
Battery charging	100	
Solar array degradation allowance	40	
<u>Core</u>		<u>425</u>
Communications	250	
ACS	100	
Power conditioning and line losses	5	
Battery charging	50	
Solar array degradation allowance	20	

Solar arrays are employed at each using site (Figure IIC-37). Based on a beginning-of-life (BOL) output of 8.15 watts/ft^2, the following array sizes are required:

Core (redundant arrays each side of structure)	52 ft^2
Assembler (each)	465 ft^2
Beam carriage (each)	120 ft^2

The assembler solar array is installed on the manipulator shoulder to prevent inadvertent contact between the array and the manipulator, as shown in Figure IIC-37. This requires two-axes motion of the array for tracking the sun. The beam pallet array is mounted on the carriage and will rotate on one axis to track the sun.

Table IIC-6 presents the electrical power systems weights.

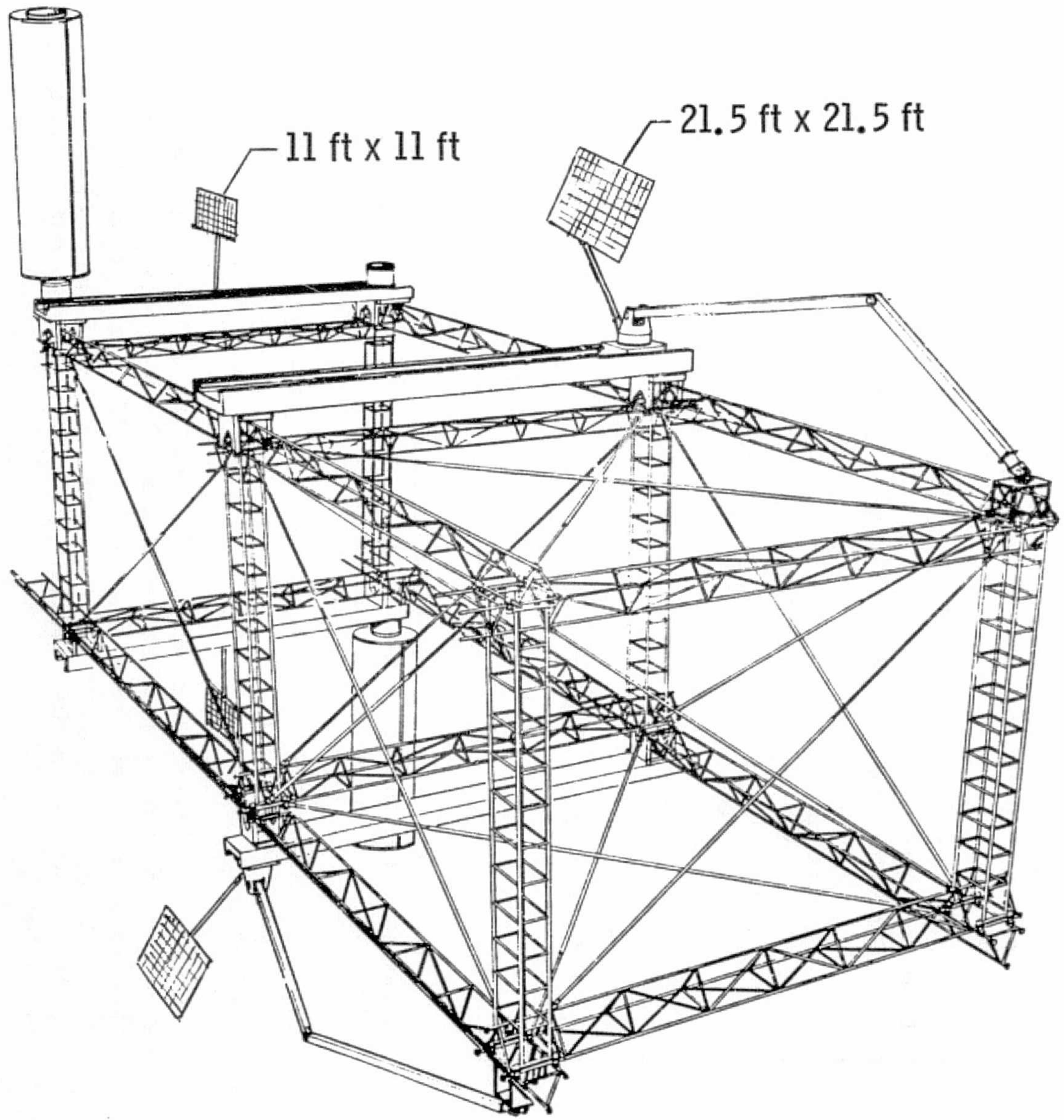


Figure IIC-37 Assembly Electrical Power Sources

Table IIC-6 Electrical Power System Weights

		Weight (lbs)
<u>Core</u>		<u>210</u>
Solar Arrays	50	
Batteries (2 at 20 amp-hr)	120	
Structure and Distribution	40	
<u>Beam Carriage</u> (pair)		<u>846</u>
Solar arrays	126	
Batteries (4 at 50 amp-hr)	560	
Structure and Distribution	160	
<u>Assembler</u> (pair)		<u>2270</u>
Solar arrays	490	
Batteries (10 at 50 amp-hr)	1400	
Structure and distribution	380	
		3326

One area of concern in the electrical power system that needs further investigation is the transfer of power and data between the assembler carriage and the subsystems in the manipulator shoulder area. The shoulder moves about 60 feet. Flexible take-up cables might be used but the effects of the space environment might be detrimental. Other methods should be investigated.

c. <u>Attitude Control System</u> - The central core section requires attitude stabilization when deployed from the Shuttle Orbiter. No problems are envisioned to provide attitude sensors and propulsion thrusters for controlling the central core and a portion of the structure. Thrusters could be initially installed in the core cross beams.

Attitude control requirements increase in magnitude as the support structure becomes larger. Table IIC-7 presents mass property data for the structure in the low-earth-orbit (LEO) and geosynchronous completed phases.

Table IIC-7 MPTS Support Structure Mass Properties

	<u>LEO</u>	<u>Geosynchronous</u>
Number of cubes	35	2709
Weight, lbs (with core and one set of assembly equipment)	53,164	1,967,436
I_{xx}, ft-lbs-sec^2	8.8×10^6	4.1×10^{10}
I_{yy}, ft-lbs-sec^2	1.5×10^7	4.2×10^{10}
I_{zz}, ft-lbs-sec^2	2.4×10^7	8.4×10^{10}

Although the structure is assembled in a spiraling fashion to maintain symmetry as much as possible, there will exist conditions where there will be a difference between the edges of the structural disc. Assuming a difference of one row of cubes between the edges, solar pressure (0.94×10^{-7} lb/ft^2) would create, in the worst case, a torque of 3.5 ft-lbs on the completed structure and 0.03 ft-lbs on the LEO structure.

Gravity gradient forces provide the potential for the greatest torques on the structure. Assuming the worst case (structure tipped 45° to the earth), the gravity gradient torque would be 336 ft-lbs on the completed structure and 29.6 ft-lbs on the LEO structure.*

To attempt to stabilize the structure to some controlled attitude would require considerable thruster propellant use and attendant resupply requirements. Moving propulsion modules radically outward (or installing new ones periodically) as the assembly progresses would create unnecessary problems. It is proposed that the structure be allowed to seek gravity gradient stabilization. This would result in an attitude where the disc would be edge-on to the earth. This attitude would also minimize solar pressure torques. The proposed communications antenna and power system solar array configurations would be compatible with this attitude.

An active attitude control system would be required to stabilize the structure from other disturbances. The requirements were investigated for stabilizing the structure from the mass momentum transfer occurring when a full beam pallet is docked to the structure.

For the worst case of docking a beam pallet at the rim of the full support structure, the resultant disc rotational velocity about the X axis (in the plane of the disc) would be about 0.16 deg/min, as determined by the equation:

$$\omega = \frac{M_{PL} \ V \ L}{M_{PL} \ L^2 + I_{xx} g}$$

where:

ω = rotational velocity, rad/sec
M_{PL} = docked payload mass = 43000 lbs

---

* $T_{G} = 3/2 \frac{GM}{r^3} \quad (I_{zz} - I_{xx}) \sin 2\theta$

V = docking velocity = 1 ft/sec assumed
L = docking radius = 1630 ft
I_{xx} = moment of inertia = 4.1×10^{10} ft-lbs-sec^2

This disturbance could be stabilized by 5-lb thrusters installed two cubes out from the central core section (edge of LEO assembled structure). This would give a thruster moment arm of 140 feet. Based on:

$$\omega t = \frac{1}{2} \frac{T}{I} t^2$$

where:

t = time, seconds
T = thruster torque = 1400 ft-lbs
I = moment of inertia of structure, including docking load = 4.5×10^{10} ft-lbs-sec^2

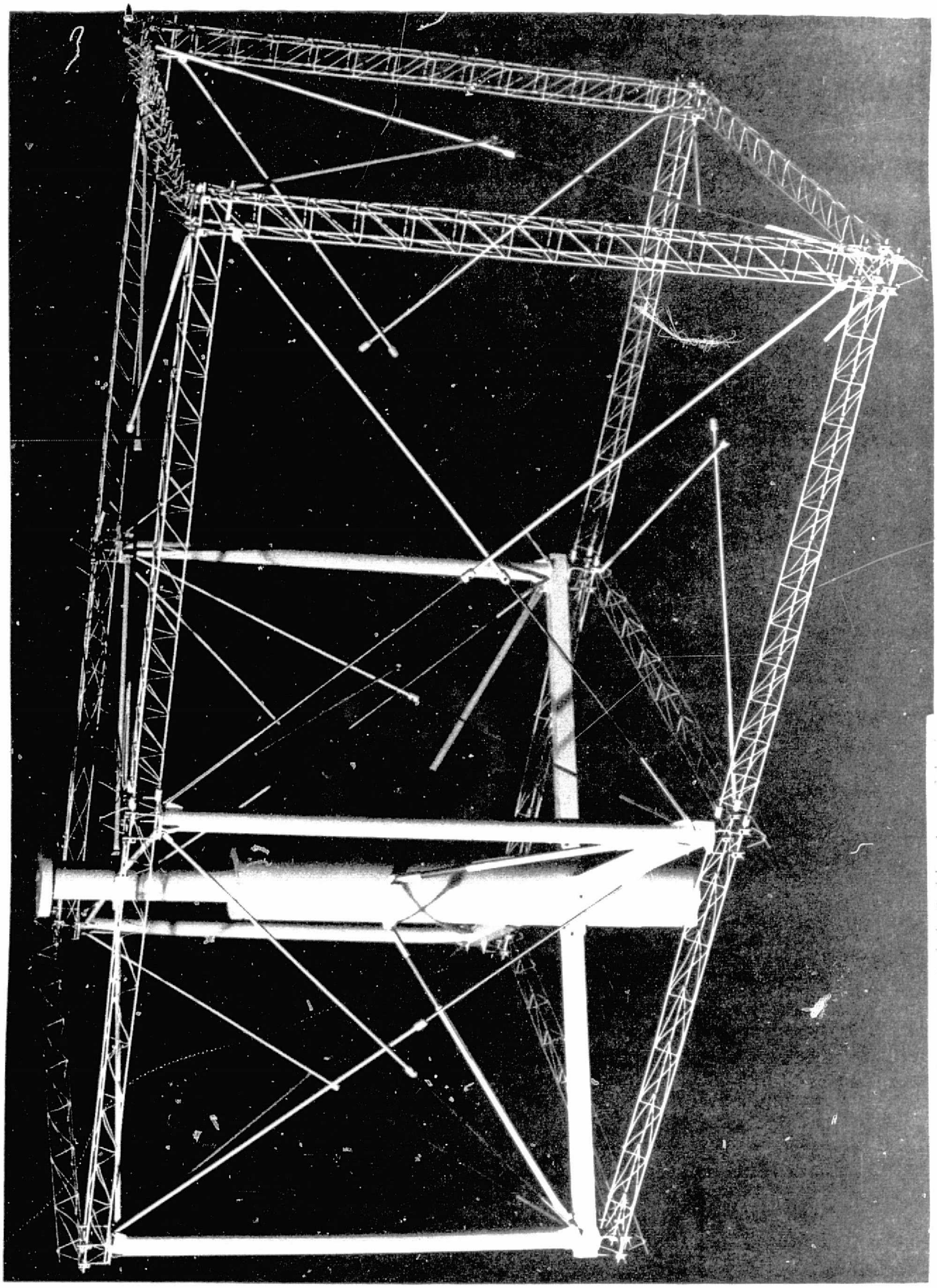
the disturbance could be nulled in 52 minutes. Assuming storable propellants with an I_{sp} of 450 seconds, about 69 lbs of propellant would be consumed. Thruster requirements would be less for payload dockings earlier in the assembly. Assuming an average propellant expenditure of 35 lbs per docking and 60 dockings, a total expenditure of 2100 lbs of propellant would accrue.

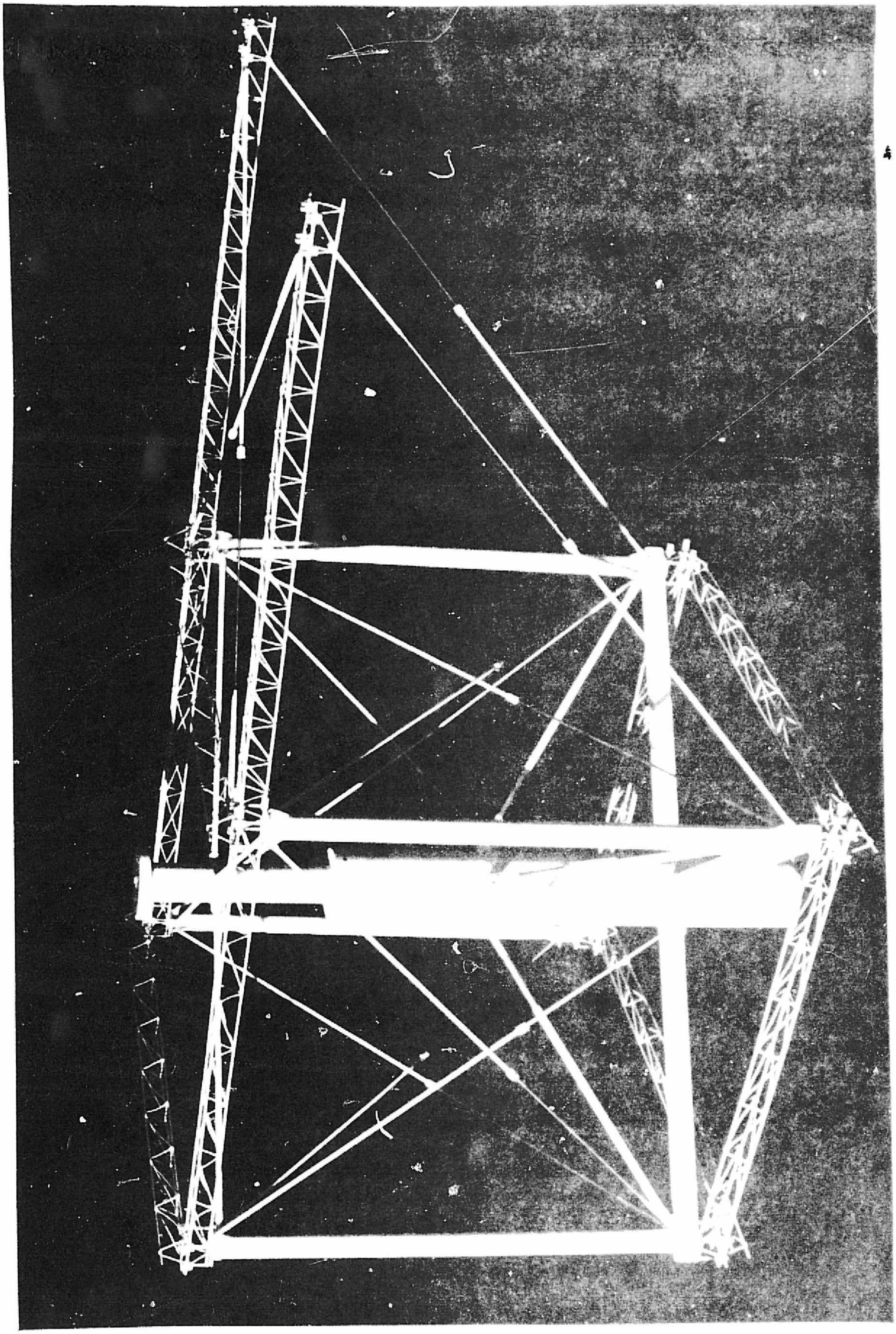
Requirements for thruster control of other minor disturbances would exist but should also be relatively small. Therefore, an attitude control system using thrusters at the edge of the LEO structure appears sufficient.

5. Structural Model

As part of Task 2 (Conceptual Design) we designed and fabricated a 1/40th scale model of the core section and one adjoining cube. Figure IIC-38 is a photograph of the model fully assembled. Figure IIC-39 shows the core section and a partially assembled cube. This model has improved our insight to the assembly procedure. In addition it has aided in establishing the reach and degrees-of-freedom requirements for the mobile assembler manipulator. The model is presently being enhanced with the addition of the mobile assembler, including two mobile carriages, a manipulator arm, and beam pallet.

This will allow demonstration of the mobile assembler carriage "walking" concept in addition to the study of physical interference problems during assembly.





D. TRADEOFFS

1. Manned vs Automated

Trade-offs were conducted to determine the most feasible approach to the assembly of the MPTS structure. We evaluated mans direct utilization in an EVA direct mode, in an indirect mode controlling the task from a nearby orbiting space station and from the ground.

Our definition of manned activities has two basic subcategories: manned direct and manned remote. Manned direct means the astronaut can physically contact the task, generally requiring EVA. Manned remote means the astronaut controls the task remotely from within the Shuttle, space station or from the ground. Within the manned remote category, there are further breakdowns of levels of manned control. For example, man can have full control of a remote task, he can have computer aiding or can act as a monitor with over-ride capability. Of course, there are further breakdowns within these categories.

In this study, we analyzed the assembly procedure for the MPTS support structure, which, in fact, is only a small portion of the total Solar Power Station. Our assembly approach is based only on the MPTS support structure. As the total power station assembly procedure evolves in the future the need for an on-site, manned station may become necessary. We based our decision to minimize manned direct activities because of the size of the beams and the repetitiveness of the tasks. The support structure contains 2709 identical 60-ft cubes, which in turn contain 32,700 beams and "X" braces. The assembly contains only three different types of components. This concept creates an assembly procedure that is extremely repetitious. Our timeline analysis showed that using the mobile assembler, it will take 9-months at 24 hours a day to complete the MPTS support structure. In addition, the beams are 60 ft long and can weigh up to 100 lbs each. Translation and alignment of thousands of beams by EVA astronauts in a MMU type vehicle has enormous logistics problems for both the astronauts and the MMU resupply. An MMU type vehicle would also have difficulty handling the inertias of the 60-ft beams.

We also considered using a free-flying teleoperator system for beam handling. Again, the logistics problems plus the problems associated with the remote control of a dynamic free-flying vehicle in and around the existing structure strongly influenced the rejection of this approach.

We have concluded that the assembly tasks as defined by our structural concept, are best accomplished with an on-site machine. This machine, the mobile assembler, has both manual and automated control functions. Since the beam alignment task requires making tolerance buildup adjustments, this phase is best accomplished by remote (via TV) manned control. The major translations of the beams, from the beam pallet to the installation site, are best controlled by preprogrammed computer control mode.

The core section of the assembly (in LEO) is unique and will depend strongly on EVA/MMU astronaut activities for initial alignment, inspection of assembled components. These tasks will take place near the Shuttle orbiter. During the HEO phase no astronaut is present for the basic assembly tasks, although in-orbit use of man will be required for the all-to-likely breakdowns that will probably occur in assembly equipment during the 9-month assembly period. In addition, we anticipate the use of astronauts in HEO for maintenance during the operational phase of the MPTS.

Procedures for the installation of wave guides assemblies, screw jacks, electrical wiring, gimbal drive units, etc. have not been evaluated. These tasks may require man directly involved in the assembly in HEO.

2. Transportation, Logistics and Cost

A very basic and important tradeoff to be addressed is that of the transportation logistics required for the assembly of the MPTS and the associated cost. The problem can be simply stated as follows: How can we best boost the antenna parts from the ground to geosynchronous orbit? What are the constraining parameters and what work has to be done in order to better assess the alternatives?

Figure IID-1 pictorially describes the problem in terms of the class of transport vehicle available to perform the task of moving the parts. Shuttle will be used from the ground to low earth orbit (LEO, nominally 160 n miles). Tugs will provide the transportation from LEO to HEO (high earth orbit, nominally geosynchronous orbit) with the possibility of using a Solar Electric Propulsion System (SEPS) from some intermediate earth orbit (IEO) to HEO to more efficiently use the boost capability of the Tug.

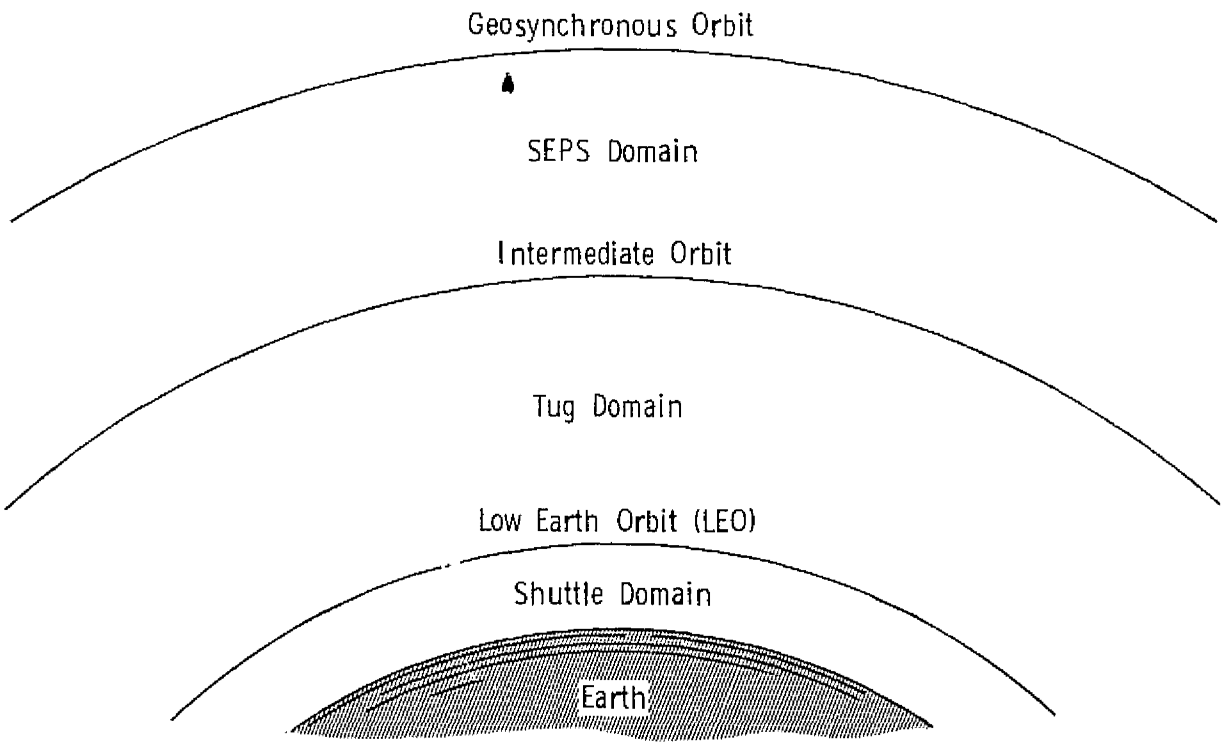


Figure IID-1 Basic Transportation Orbital Regimes

SEPS is described in detail elsewere[1], but some of its more important parameters relating to this problem are summarized in Figure IID-2. The parameters which impact assembly transportation are SEPS lifetime (700 days) and its performance capabilities in terms of the mass that can be transported in some period of time. Problem areas such as sun occultation (both by the earth and by the payload to be moved) and the overall control limitations of SEPS are mentioned as items to be studied at a later time.

Tug has the capability to boost the parts to HEO, but due to the large number of Tugs needed, cost becomes a significant factor. The following is a discussion of the various methods to use each boost vehicle and the rationale behind the finalized concept.

Variations of payload weights that can be packaged into the Shuttle cargo bay will be factored into the following discussion of Tug options featuring the performance characteristics of the full capability Tug.

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[1] "Concept Definition and Systems Analysis Study for a Solar Electric Propulsion Stage," Rockwell International, SD74-SA-0176-1.

Intended Use: Increase Tug Payload 50 to 100% and On-Orbit Servicing

Developmental Status: Concept Development, Components Tested, Not Presently Baselined for Shuttle

Size (Not Deployed) 10 x 15 ft

Stage Weight - 2590 lb

Hg Propellant Weight - 3270 lb

Solar Array Procedures - 25 kw

Thrusters - (9) Ion, Total Thrust = 0.206 lb, I_{SP} = 3000 sec

Typical Mission - Tug to IEO, SEPS to HEO

Mission Durations: 1 to 2 Months (Or More)

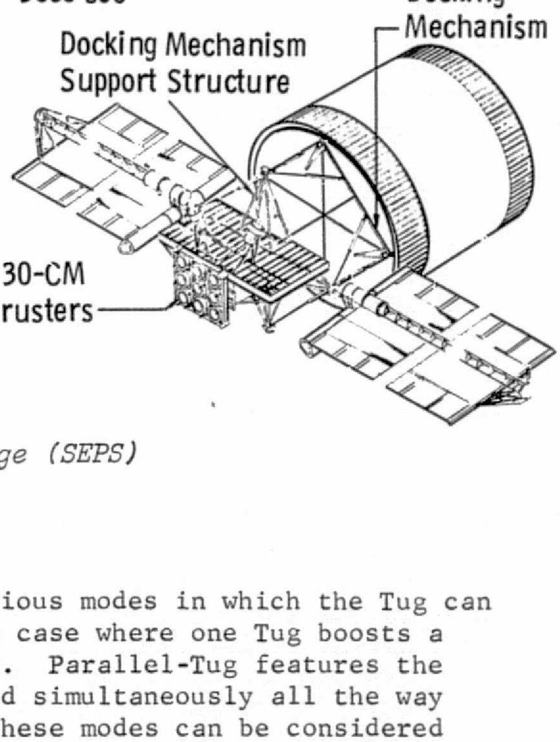


Figure IID-2 Solar Electric Propulsion Stage (SEPS)

Figure IID-3 describes the various modes in which the Tug can be used. Single Tug represents the case where one Tug boosts a payload up to some desired altitude. Parallel-Tug features the case where two Tugs thrust a payload simultaneously all the way to its desired altitude. Both of these modes can be considered with Tugs reused (returned to LEO) or Tugs expendable (the latter being the higher performing case).

The tandem combination features two Tugs (T1 and T2) where T1 boosts T2 and the payload up to some intermediate altitude at which time T2 is activated and continues boosting the payload while T1 has enough fuel to return to LEO. T2 has enough fuel left to boost the payload to the desired orbit and still return to LEO also.

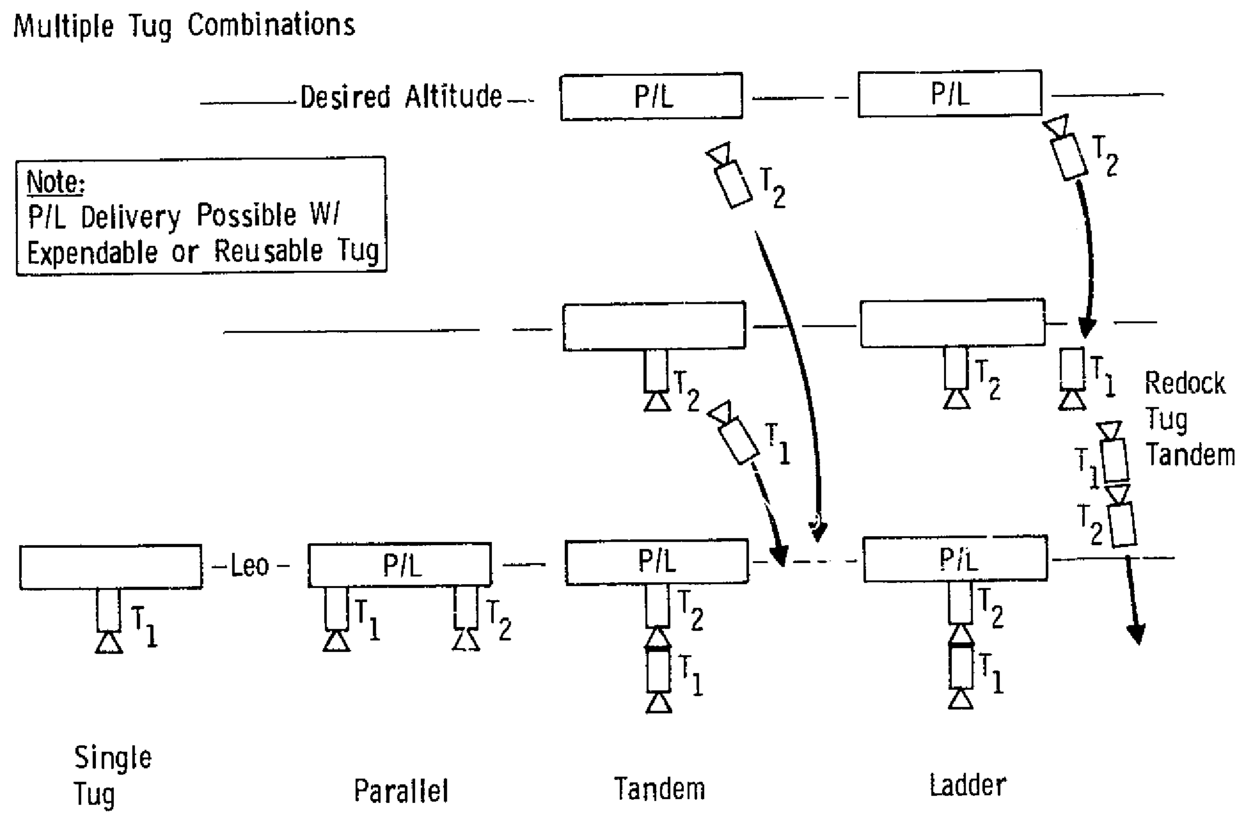


Figure IID-3 Tug Combinations for Payload Docking

The ladder approach is similar to the tandem approach except that T1 remains at some intermediate orbit with sufficient fuel to carry itself and the expended T2 (which has boosted the payload to HEO and come back to the T1 altitude) back to LEO. Single, parallel, and tandem Tugs can be considered with the Tugs assumed expendable, which yields substantially higher performance. One must, of course, recognize the higher costs to be suffered in the expendable case in terms of the higher Tug and Shuttle prices if they are not to be reused. Table IID-1 below demonstrates the performance of the various Tug modes in terms of the payload that can be delivered from 160 n miles (Shuttle nominal orbit) to geosynchronous altitude (19,300 n miles) or to various intermediate altitudes such as 10,000, 12,500, 15,000, and 17,500 n miles.

Table IID-1 Delivery Capability of Various Tug Combinations (lbs)

From 160 n miles to: Altitude (n miles)	Tandem 2 Stage Delivery	Ladder Tandem 2 Stage Delivery	2 Stage Expendable	One Stage Delivery	One Stage Expendable
8000	71000	72000	85000	28000	42000
10000	65000	66000	77000	25000	38000
12500	54000	57500	70000	20000	35000
15000	47000	52000	65500	19000	31000
17500	45600	51000	62500	18000	28000
18200	44000	49000	62000	17000	27000
Geosynchronous	35000	41000	53000	8000	26000
Weight of Unfueled TUGS = 6050 lbs Isp of TUG Propellant = 450 sec Weight of TUG Propellant = 50190 lbs					

Calculations of the total transportation cost to boost 5.8×10^6 lbs of payload into geosynchronous altitude resulted in the information depicted in Figure IID-4. The two-Tug-ladder reusable mode is shown to be the best approach from a cost standpoint, assuming \$5.9M per expendable Tug and \$1M for a reusable Tug cost. It should be mentioned that the numbers justifying the two Tug ladder reusable scheme are only slightly higher than the two-Tug-tandem expendable approach and should the assumed reusable or expendable Tug costs change, the costs should be recalculated to be sure the lowest approach is considered.

The three curves shown are parameterized in one sense to demonstrate the cost difference as a function of Shuttle payload capability. As one can boost more in each Shuttle, the total number of shuttles needed decreases. Shuttle costs are assumed to be \$10M each.

The other parameter to be varied is the intermediate altitude at which SEPS can relieve Tug and perform the final transportation phase. The right hand portion of the curve represents the total costs when the intermediate altitude is geosynchronous (i.e., no SEPS are used). One can see that 290 Tugs are needed and no SEPS. As the SEPS begins being considered, one notices a substantial savings with the optimum altitude in the 15,000 to 17,000 n mile range. An apparent \$0.6B to \$0.7B saving seems attainable, even though there are a few negative factors with this approach.

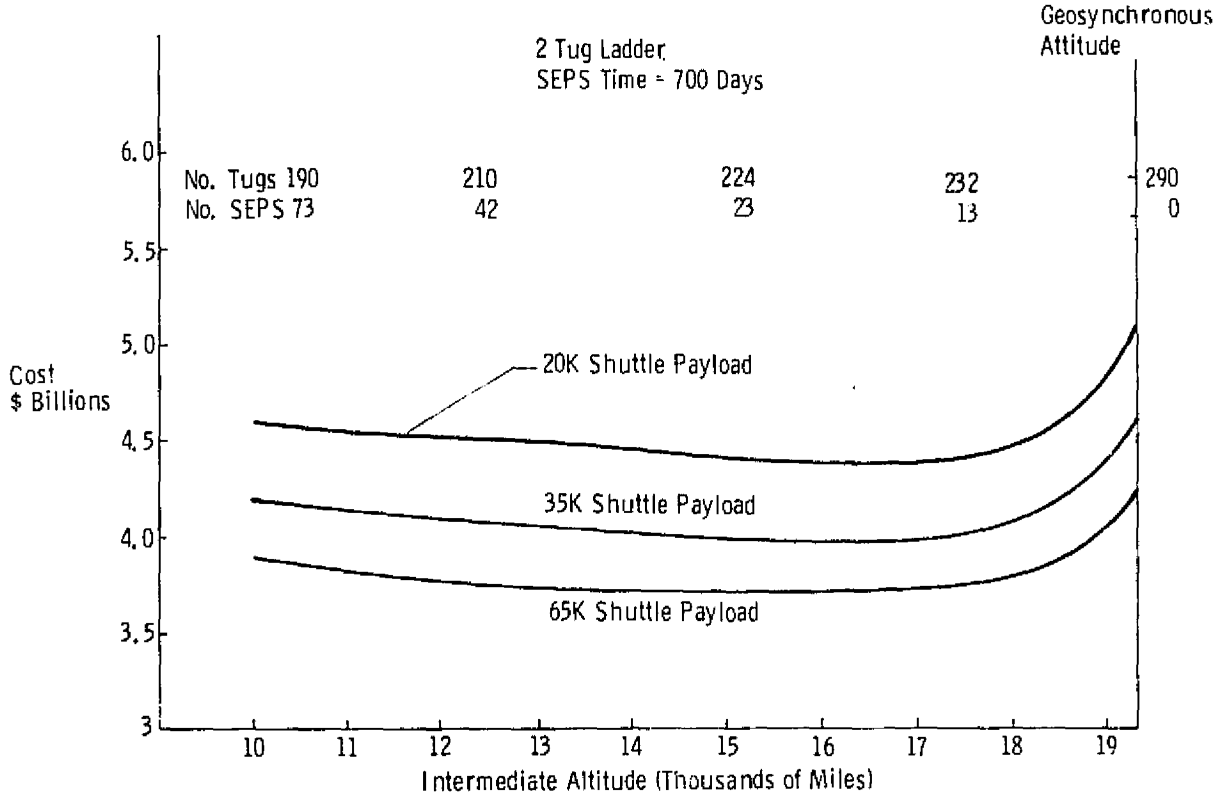


Figure IID-4 Transportation Costs to Boost 5.8 x 10^6 lbs to Geosynchronous Orbit for Va ying Intermediate Altitudes and Shuttle Payloads

These curves all assume that SEPS total lifetime of 700 days will be used, which effectively adds 700 days to the total transportation time. Tugs can deliver the payloads in a matter of hours. As an example, if 400 Shuttle flights are needed and one Shuttle is launched every day, it will take 400 days to get all the payload to LEO. The last payload can then be delivered to HEO on the 400th day using Tug. Using SEPS, the last payload would not get to HEO until the 1100th day.

A number of factors must be considered. If Shuttles cannot be launched every day and may be launched only every third or sixth day, the Shuttle transportation time may be as high as 2400 days. The extra 700 days for SEPS becomes lower percentage-wise and may be more attractive when considering the cost savings. SEPS also still has practical performance problems that must be studied. While large payloads can theoretically be boosted by SEPS (e.g., hundreds of thousands of pounds), the occultation of SEPS solar panels by the very large structure (it is effectively transparent, however), may limit the SEPS power output. The ability of SEPS to provide control torques to so large a structure must be analyzed, the analysis as to the optimum time to perform plane and phase changes must be performed, and the use of SEPS in a mode requiring less than 700 days may be attractive and should be investigated.

With these factors in mind, the latter approach was studied to show the advantages and disadvantages of using SEPS for less than its 700 days lifetime. Effectively, one sacrifices some cost savings for a shorter total transport time.

Table IID-2 is used to demonstrate this example. It shows total transportation time and cost to deliver 5.8 x 10^6 lbs to geosynchronous orbit, for Shuttle launches once a day and once every three days versus Tugs only and Tugs and SEPS combinations with different SEPS lifetimes. The reference intermediate orbit is assumed to be 15,000 n miles. Shuttle costs were assumed to be $10 million; Tugs $1 million and SEPS $10 million.

Table IID-2 Total Transportation Time vs Cost to Boost 5.8 x 10^6 lbs to Geosynchronous Orbit

	SEPs TRAVEL TIME	NO. OF TUG FLIGHTS	NO. OF SHUTTLE FLIGHTS	NO. OF SEPs FLIGHTS	ELAPSED TIME FROM 1st TO LAST SHUTTLE FLIGHT	TOTAL TRANSPORTATION TIME	TOTAL TRANSPORTATION COST, $BILLIONS
SHUTTLE LAUNCH EVERY DAY	(No SEPs)	290	423	0	423	423	4.5
	700	228	357	23	357	1057	4.0
	350	234	363	46	363	713	4.34
SHUTTLE LAUNCH EVERY THREE DAYS	(No SEPs)	290	423	0	1269	1269	4.5
	700	228	357	23	1071	1771	4.0
	350	230	359	28	1077	1427	4.05

NOTE: All SEPs usage assumes 15,000 nautical miles intermediate earth orbit.

The table demonstrates in the top portion, in which Shuttles are launched every day, that 423 Shuttles are needed when no SEPS are used (more Shuttles are needed because more Tugs are used). At one flight per day all the payload can be boosted to LEO in 423 with the Tugs boosting the final payload to HEO within the same day as the last Shuttle flight. When SEPS are used in their 700-day mode, $0.5B can be saved in either the Shuttle launch every day or every third day cases. The difference is the 700 days added to the 357 days (one for each Shuttle flight, reduced now because there are less Tugs) and/or 700 days added to the 357 x 3 days for Shuttle flights every third day.

C. 2

When SEPS are used in a 350-day mode, they cannot boost as much payload so more of them are needed to do the task. Thus, while 350 days are saved totally in the top example (one Shuttle flight per day), only $0.16B is saved. An interesting point is made from the bottom example in which Shuttles are launched every third day. Transport time of 350 days is still being gained but since Shuttle times are up to 1071 days, the SEPS now can be reused, still using them for 350 days per trip. A substantial savings can be realized in this reuse capability. As is shown, the cost savings is $0.45B from the $4.5B total without SEPS.

The generalized conclusion is that by considering SEPS in conjunction with other mission constraints, such as how often a Shuttle can be used, may offer substantial savings at not too great a percentage loss in transport time. This should justify the more detailed study of the SEPS performance capability problem areas as well as the added justification that the SEPS cost has a good chance of being reduced more drastically than Shuttle or Tugs over the next few decades.

E. PROCEDURES AND TECHNIQUES (TASK 5)

1. Phase 1 - LEO Assembly

The MPTS support structure is assembled in two phases. In the first phase, a core structure is constructed while attached to the Shuttle Orbiter. The Shuttle Remote Manipulator System (RMS) and manned extravehicular activity (EVA) is used in the process. Accuracy of alignment of the central core is assured by two methods: (1) ground erection and alignment with precision tools, and (2) verification of alignment (and necessary adjustments) when assembled at the orbiter, through optical sightings by EVA crewmen. Mobile assemblers and communications and attitude stabilization equipment are installed and the assembly is deployed. Figure IIE-1 shows the steps in constructing the core section.

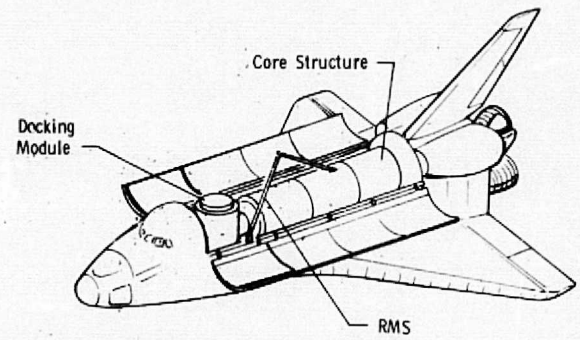
After the core section is deployed, the assemblers continue building cubes until the structure is a rectangle 5 cubes by 7 cubes in size (including the gimbal support structure). This center segment of the antenna structure is then boosted to geosynchronous orbit. The detailed steps and timelines for Phase 1 are outlined in Table IIE-1.

In Phase 2, the support structure assembly is completed through the addition of structure cubes until the required size and shape is attained. The additional structure elements, contained in beam pallets, are transported to orbit by Shuttle and Tug vehicles, and docked to the structure.

2. Phase 2 - HEO Assembly

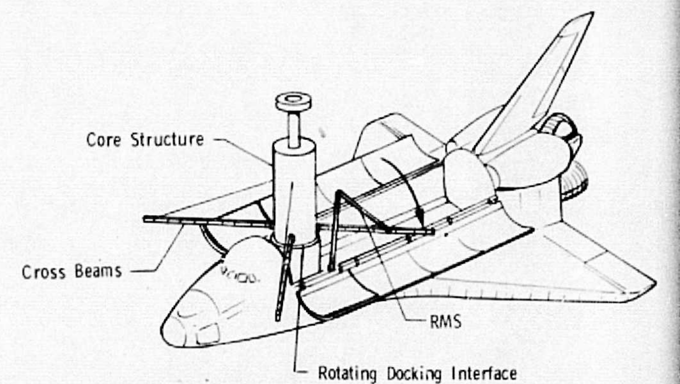
In the second phase, we assemble the remaining 2670 cubes in geosynchronous orbit. Figure IIE-2 presents views of the sequence for assembling an in-line cube (full-cube). It should be noted that although no astronauts are shown in these figures, EVA astronauts will be required when breakdowns or malfunctions occur in the assembly equipment or associated subsystems. At present, our concept for beam alignment and fastening during assembly does not include an astronaut, but we have not completely verified that this can be done remotely. In addition, the operational maintenance of the completed power station very likely will require EVA astronauts.

Assembly procedures and times are outlined in Tables IIE-2 and IIE-3. The following nomenclature is used in the procedures. Figures IIE-3 and IIE-4 present views of the assembled cubes and the element definition symbols.



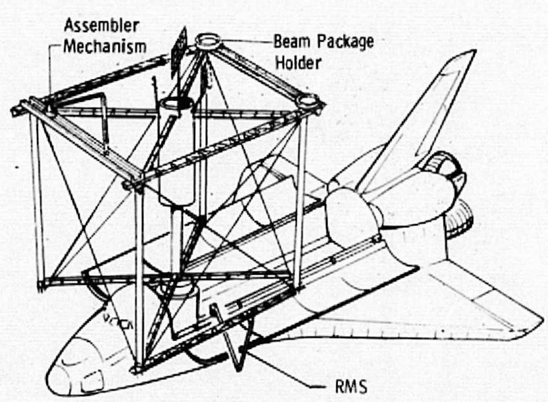
Step 1. Deploy and Dock Core (Shuttle No. 1)

The first phase in the total construction process is to assemble the middle 60-ft cube of the microwave antenna. The first Shuttle flight contains the basic core structure with folding alignment and support members, 12 beam members with X braces, and two sets of mobile assemblies and beam holders. During Step 1, the center core is extracted from the Cargo Bay, positioned and docked on the Shuttle docking module with the RMS.



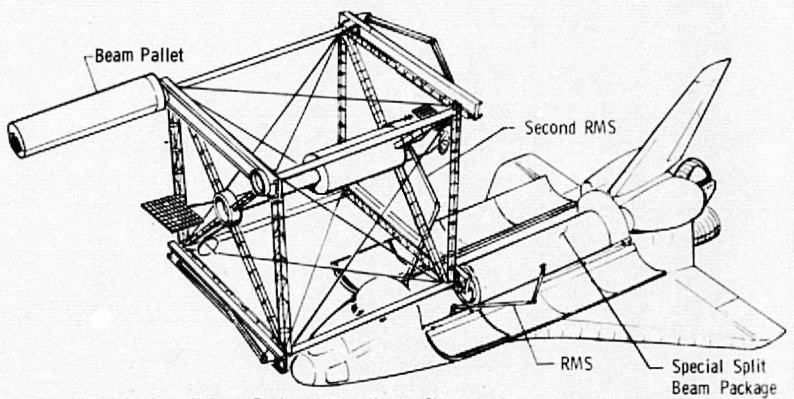
Step 2. Extend Cross Braces (Shuttle No. 1)

During Step 2 of the antenna core assembly, the alignment beams are unfolded from besid the core structure and the tension rods are positioned. Each beam is checked and vernier adjustments are made for precise alignment. A rotary docking interface is required at the dockin port since the RMS cannot reach completely around the core structure. This rotary docking ring can be an unpowered slip ring since the RMS can position the core beams by pulling the structure around.



Step 5. Rotate Assembly, Emplace Upper Beams, and Install Assembler Equipment (Shuttle No. 1)

The second set of assembler equipment is installed on the bottom side.



Step 6. Install Beam Packages (Shuttle No. 2)

The second Shuttle flight contains two beam packages. These are nonstandard in that they are split longitudinally so that a package can be placed on each side of the 60-ft core cube. The beam packages are 60-ft long and fill the cargo bay. This eliminates the use of the docking module kit. A second RMS is used to capture and stabilize the antenna core while the primary RMS places the beam package in the assembler beam package holders. This task is repeated on the opposite side. The antenna core is now ready to self-erect additional 60-ft cubes.

Figure IIE-1 Core Structure Assembly Steps

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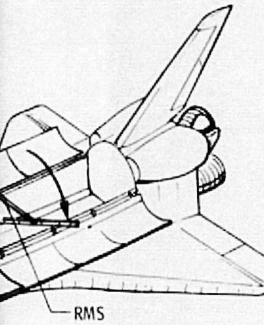

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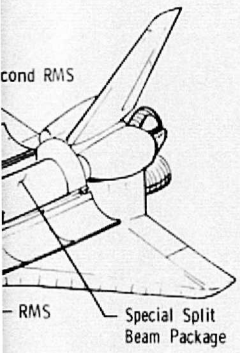

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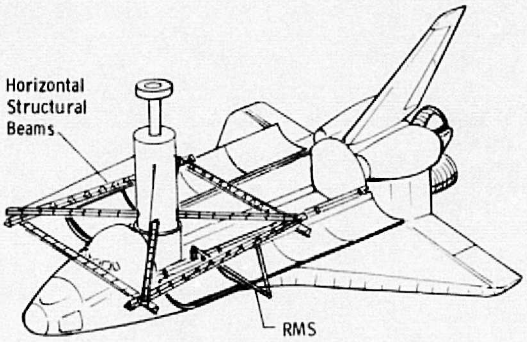
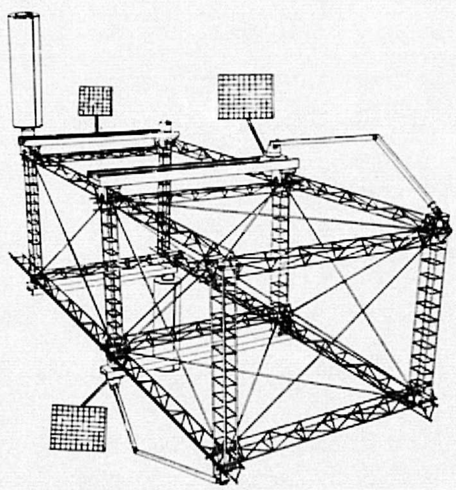


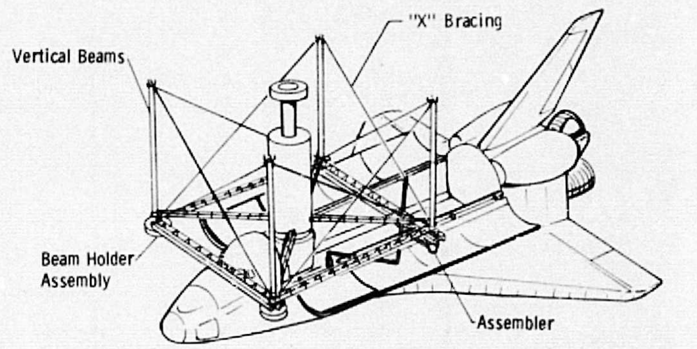
Step 3. Emplace Horizontal Beams (Shuttle No. 1)

Step 3 consists of extracting the beams from the cargo bay and placing these lower triangular beams onto the alignment beams and welding them in place, one at a time.



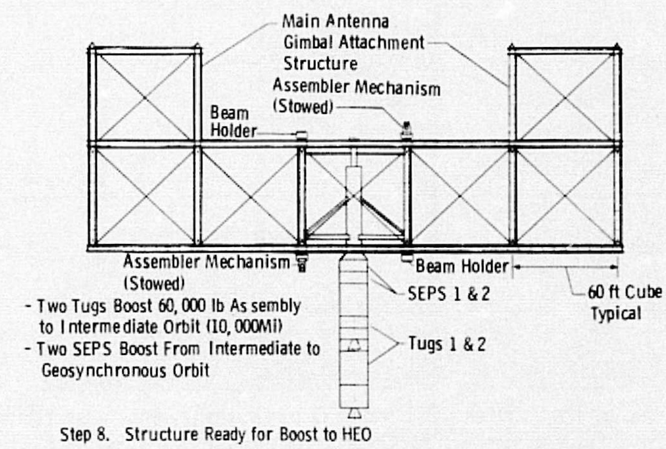
Step 7. Construct LEO Structure

Thirty eight more cubes are constructed using the assemblers. When this structure is completed, the assembly equipment is stowed and the structure is readied for boost.



Step 4. Emplace Vertical Beams, Cross Bracing, and Assembler Equipment (Shuttle No. 1)

Step 4 consists of placing and welding the vertical beams plus placement of a set of assembler equipment. The vertical beams used for this core segment are special tubular members which also contain two adjustable tension tubes, hinged from the top. Each vertical beam is placed on its adjoining corner receptacle and welded. The tension tube is extended in its plane and welded on the unattached end. The beam is then aligned in that plane with the RMS and the pyro-pin is activated within the telescoping segment of the tension tube to lock the tension tube in that position, which in turn holds the beam in alignment. This sequence is repeated for each vertical beam. One mobile assembler and one mobile beam package holder are placed on their receptacles on the lower core structure.



Step 8. Structure Ready for Boost to HEO

The 39 cube structure is readied for boost by docking two Tugs and two SEPS. This assembly will then be boosted to intermediate orbit with two Tugs. The Tugs will return and the two SEPS will boost the assembly to high earth orbit.

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Table IIE-1 Core Assembly Procedures

	Primary Function		Secondary Function	Task Time (hours)	Mission Time	Orbiter Payload Wgt.	Component Wgt. & Size	Task Control Center Location	Support Equip. needed
			First Shuttle Orbiter		0.00, Day 1				
1.0	Deploy and Dock Core	1.1	Maneuver core structure to docking port with RMS	0.5	0.5	15K	10'x40' 10,000 lbs	Orbiter	RMS and docking module
		1.2	Align & attach	0.1	0.6	-	10K	"	"
		1.3	Extend telescoping section with RMS	0.2	0.8	-	TBD Force	"	"
2.0	Extend Cross Braces	2.1	Unfold four "X" beams & lock with RMS	0.5	1.3	-	"	"	"
3.0	Emplace Horizontal Beams	3.1	Extract, maneuver and emplace 60-ft lower horizontal beam	0.3	1.6	-	3'x60', 150 lbs	"	"
		3.2	Align beam ends & attach "X" braces	0.2	1.8	-	"	"	RMS and EVA
		3.3	Rotate assembly with RMS	0.2	2.0	-	-	"	Rotating docking ring and RMS
		3.4	Repeat 3.2 and 3.3 for three more lower beams	2.1	4.1	-	3'x60', 150 lbs	"	RMS
4.0	Emplace Vertical Beams and Assembler Equipment	4.1	Extract, maneuver & emplace four 56-ft vertical beams on corners	2.8	6.9	-	2'x56', 250 lbs	"	"
		4.2	Extend, align & attach cross bracing between vertical and horizontal beams	4.0	10.9	-	4"x70', 50 lbs	"	RMS and EVA
		4.3	Emplace pair of assembler and beam holder mechanisms	2.0	12.9	-	2'x4'x60' 2500 lbs each	"	RMS
					Day 2				
		4.4	Activate and checkout assembly ACS	0.5	24.5	-	-	"	-
5.0	Rotate Assembly and Emplace Upper Beams	5.1	Undock assembly and deploy communications antennas and solar array on one side	0.1	24.6	-	11K	"	-
		5.2	Maneuver Shuttle to dock on opposite side of assembly	2.0	26.6	-	-	"	-
		5.3	Lock Shuttle to assembly	0.5	27.1	-	11K	"	-
		5.4	Extract, maneuver & emplace 60-ft lower horizontal beams (4 reqd)	1.2	28.3	-	3'x60', 90 lbs	"	RMS
		5.5	Align beam ends & attach to verticals	0.8	29.1	-	"	"	RMS and EVA
		5.6	Rotate assembly with RMS	0.8	29.9	-	-	"	RMS

Table IIE-1 (Continued)

Primary Function	Secondary Function	Task Time (hours)	Mission Time	Orbiter Payload Wgt.	Component Wgt. & Size	Task Control Center Location	Support Equip. needed
	First Shuttle Orbiter						
5.0 Rotate Assembly and Emplace Upper Beams (Cont'd)	5.7 Attach "X" bracing from locking ring to structure corners (4 reqd)	1.6	31.5	-	4"x70', 30 lbs	Orbiter	RMS and rotating docking ring
	5.8 Emplace pair of assembler and beam holder mechanisms	2.0	33.5	-	3'x4'x60' 2500 lbs each	"	RMS
	5.9 Undock, deploy communications antennas and solar array, and verify ACS	0.5	34.0	-	12K	"	Docking port
	5.10 Activate assembler electrical power and checkout all mechanisms	1.5	35.5	-	-	"	-
			Day 3				
	Second Shuttle Orbiter						
6.0 Install beam packages	6.1 Rendezvous, capture and position antenna core assembly with RMS	2.0	50.0				
	6.2 Extract, maneuver & emplace first beam package (half) on assembler beam holder	0.8	50.8	32K	11K	"	(2) RMS
	6.3 Rotate assembly 180° with RMS and stabilize	2.5	53.3	"	33K	"	" "
	6.4 Repate 6.2 for second beam package	0.8	54.1	"	44K	"	" "
	6.5 Attach second RMS to antenna assembly	0.2	54.3	"	-	"	" "
	6.6 Activate & checkout mobile assemblers (elect. power)	4.0	58.3	32K	44K	"	" "
	6.7 Activate assembly ACS, release & verify stabilization & beam holders	0.5	58.8	"	"	"	" "
	6.8 Extract beams from beam holders & begin assembly process. (ground control - check comm. network & timeline)	0.5	59.3	-	-	Ground	Possible EOTS
			Day 4				
7.0 Construct LEO Structure	7.1 Build 38 more cubes (see Tables IIE-2 and IIE-3)	86.4	182.4	-	-	"	" "
	7.2 Shuttle crew monitor assembly & alignment of beams, and possible EVA repair	-	-	-	-	-	EVA
	7.3 Stow antennas and solar arrays and deactivate assembler and holder mechanisms for TUG boost	2.0	184.4	-	-	-	-
			Day 9				

Table IIE-1 (Concluded)

Primary Function		Secondary Function	Task Time (hours)	Mission Time	Orbiter Payload Wgt.	Component Wgt. & Size	Task Control Center Location	Support Equip. needed
		Third Shuttle						
8.0 Prepare Structure for Boost to HEO	8.1	Rendezvous Shuttle with antenna assembly	2.0	194.0	65K	60K TUG 5K SEPS	Orbiter	(1) RMS
	8.2	Extract TUG/SEPS from cargo bay with RMS	2.0	196.0	"	65K	"	"
	8.3	Checkout TUG/SEPS	0.5	196.5	"	"	"	"
	8.4	Position & release TUG/SEPS	0.2	196.7	"	"	"	"
	8.5	Maneuver TUG/SEPS to antenna assembly	0.5	197.2	-	-	Orbiter & Ground	-
	8.6	Align & dock TUG/SEPS to upper antenna assembly docking ring	0.3	197.5	-	-	" "	-
	8.7	Undock TUG & maneuver to lower docking ring	0.4	197.9	-	-	" "	-
	8.8	Dock TUG to lower docking ring	0.3	198.2	-	-	" "	-
		Fourth Shuttle						
	8.9	Rendezvous Shuttle with antenna assembly	2.0	200.2	65K	60K TUG 5K SEPS	Orbiter	(1) RMS
	8.10	Extract second TUG/SEPS from cargo bay with RMS	2.0	202.2	"	65K	"	"
	8.11	Checkout TUG/SEPS	0.5	202.7	"	"	"	"
	8.12	Position & Release TUG/SEPS	0.2	202.9	"	"	"	"
	8.13	Maneuver TUG/SEPS to upper docking area on antenna	0.5	203.4	-	-	Orbiter & Ground	-
	8.14	Align & dock SEPS to aft end of first SEPS docked on antenna	0.3	203.7	-	-	" "	-
	8.15	Maneuver TUG to lower docking area on antenna	0.4	204.1	-	-	" "	-
	8.16	Align & dock second TUG to aft end of first TUG	0.3	204.4	-	-	" "	-
	8.17	Verify operational integrity of all systems & deactivate for boost						
	8.18	Align & initiate TUG boost						

Day 10

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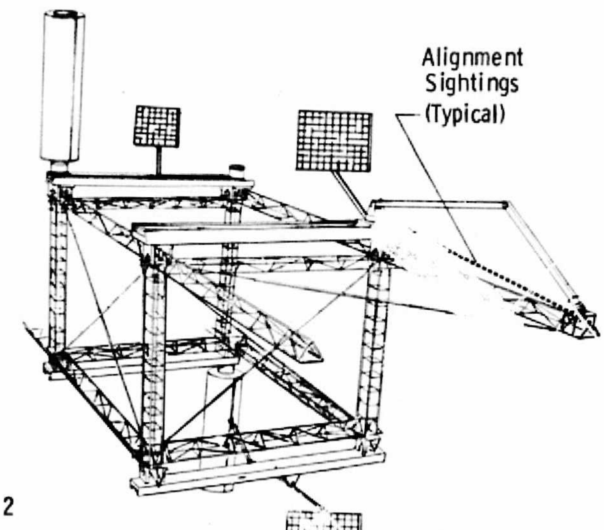
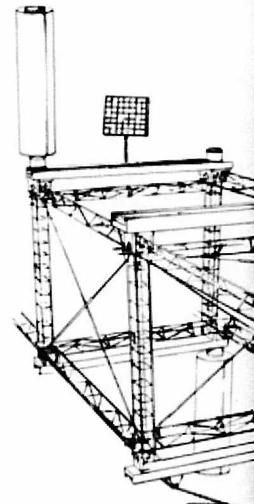


Steps 1 and 2

This series of assembly sequences shows the typical beam placement tasks required to complete each 60-ft structural cube. The first triangular beam is placed on the end of the previously constructed beam end and welded in place. The horizontal telescoping tension tube is extended to the opposite corner and welded. The manipulator positions the beam to align in the plane of the tension tube. The pyro-pins are activated to lock the telescoping tension tube segment and in turn holds the beam in place. This procedure is repeated for the vertical tension tube. By locking the tubes in this manner, proper alignment of the beam is assured. This is repeated for the second horizontal cap beam.



Step 3

Step 3 places the crossing, triangular b
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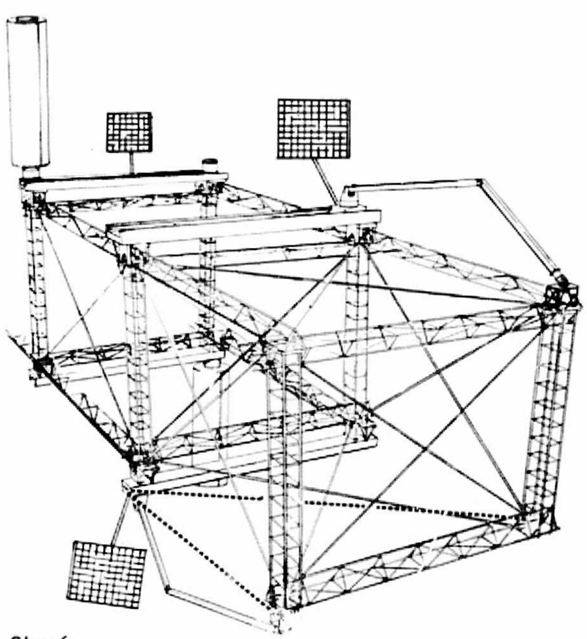
with alignment pins. It will then move to th
along the other horizontal beam. Once in p



Step 6

Step 6 places the lower triangular cross beam at the lower ends of the vertical beams. It is aligned and welded on each end.

Steps 7 and

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Figure IIE-2 Structural Section Assembly Steps

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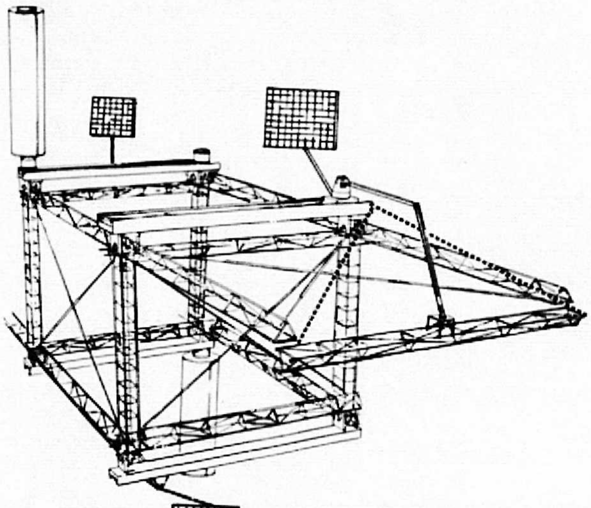
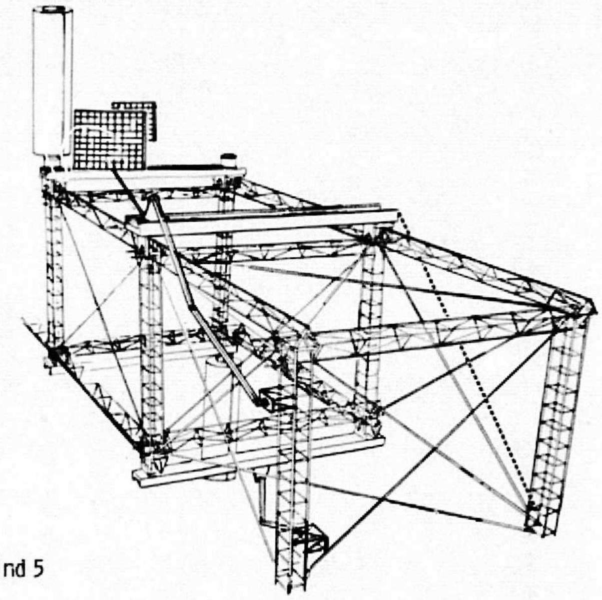
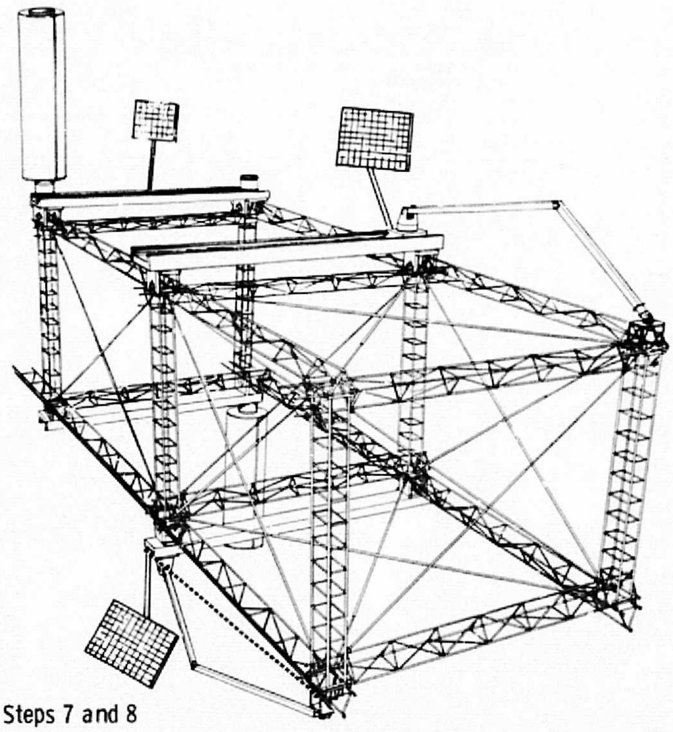


places the crossing, triangular beam at the outer beam ends. This beam is aligned in place. The manipulator arm will hold the beam in the center and locate one end nent pins. It will then move to the other end of the beam and locate the cross beam other horizontal beam. Once in position both ends will be welded in place.



Step 4 and 5

These steps places the two vertical (square) beams on the cube end. Their ends are welded in place and again aligned with the cross braces. Both the top and bottom manipulators will be used for this task.



Steps 7 and 8

The two lower horizontal beams are now put into place. They are individually abutted to the adjoining beam end and welded. They are aligned on the outer end and also welded in place. The lower tension tubes are locked as the manipulator locates the end of each horizontal beam.

NOMENCLATURE

P	Corner joint positions
A-T	Top assembler (base from P-1 to P-2)
A-B	Bottom assembler (base from P-3 to P-4)
B-T	Top beam pallet (assumed in line with P-1)
B-B	Bottom beam pallent (assumed in line with P-4)
A	Automatic action after ground command
M	Manual remote control from ground (TV assist)
T	Triangular beams
	T - nominal 56.2 ft beam
	T_x - extended beam to span beyond 56.2 ft centers
R	Rectangular beams
C	Cross braces (A indicates brace for adjacent cube)

Cameras designated by assembler end position, e.g., A-T-1 is camera on A-T at P-1 end.

The tables also designate those tasks that would be manually directed (M) by remote control by a ground operator and those tasks that are automated (A) and directed by the on-board computer.

The discrete task times are analytical estimates. Two of the tasks were simulated and are discussed in Chapter II. The simulations verified several of these analytical estimates and in one case, the simulated task time was approximately 15% less than the estimates.

Assembly of the in-line cubes (full-cube) is estimated to take 2 hours and 51 minutes. Assembly of the cubes in the quadrants (partial-cube) is estimated to take 2 hours and 11 minutes.

Total assembly time for the entire antenna support structure will be about 259 days, assuming full time activity using one pair of assemblers, excluding the time for transport of the partial assembly to geosynchronous orbit. Obviously, additional pairs of assemblers, constructing cubes, will decrease the total assembly time proportionally.

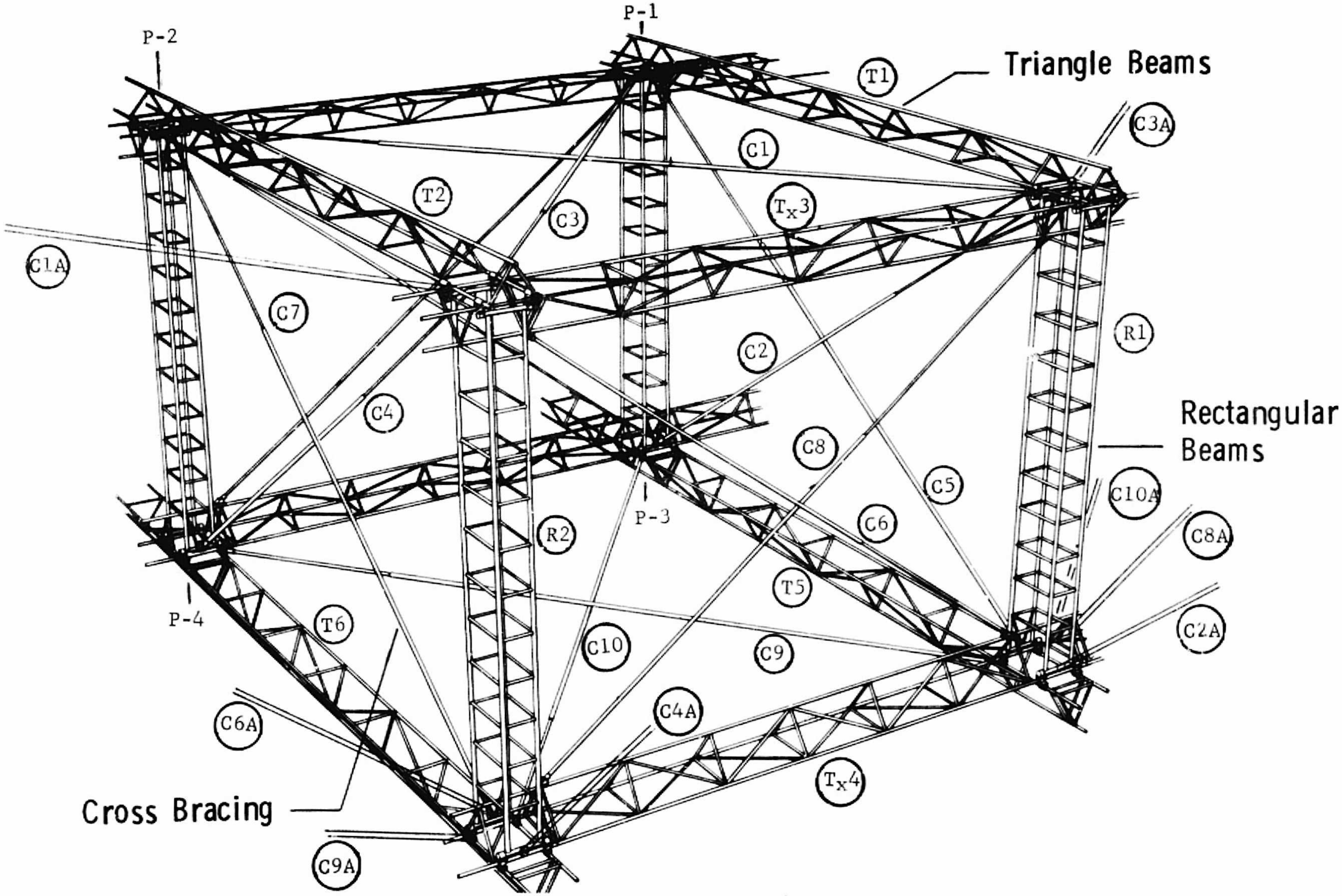


Figure IIE-3 Full-Cube Structure

Table IIE-2 Full-Cube Assembly Procedures

Primary Function		Secondary Function	Mode	Task Time (Minutes)
1.0 Install Beam T1	1.1	Position A-T shoulder at P-1	A	1
	1.2	Position A-T end effector at B-T	A	2
	1.3	Rotate B-T to expose proper quadrant	A	
	1.4	Extract T1	M	2
	1.5	Open T1	M	1
	1.6	Lock opened T1 (pyro)	A	-
	1.7	Extract C1 and attach to T1	M	3
	1.8	Extract C3A (for next cube) and attach to T1	M	3
	1.9	Separate T1 from BT	M	1
	1.10	Position T1 at P-1	A	2
	1.11	Insert T1 into P-1	M	1
	1.12	Fasten T1 in place	A	-
	1.13	Extend C2 to opposite corner (C2 initially attached with last cube)	M	2
	1.14	Weld C2 end	A	-
	1.15	Extend C1 to opposite corner	M	2
	1.16	Weld C1 end	A	-
	1.17	A-T-1 and A-B-3 cameras aligned with core bench marks	A	1
	1.18	Point A-T-1 and A-B-3 cameras for T1 alignment and leveling	A	-
	1.19	A-T manipulate T1 to alignment	M	2
	1.20	Lock C1 (pyro)	A	-
	1.21	Lock C2 (pyro)	A	-
T1 Installed - Task Time			23 minutes	
Total Accumulated Time			23 minutes	
2.0 Install Beam T2	2.1	Position A-T end effector at B-T	A	2
	2.2	Extract T2	M	2
	2.3	Open T2	M	1
	2.4	Lock opened T2 (pyro)	A	-
	2.5	Extract C3 and attach to T2	M	3
	2.6	Extract CIA and attach to T2	M	3
	2.7	Separate T2 from B-T	M	1
	2.8	Position T2 at P-2 (shoulder travel)	A	3
	2.9	Insert T2 into P-2	M	1
	2.10	Fasten T2 in place	A	-
	2.11	Extend C4 to opposite corner	M	2
	2.12	Weld C4 end	A	-
	2.13	Extend C3 to opposite corner	M	2
	2.14	Weld C3 end	A	-
	2.15	A-T-2 and A-B-4 camera aligned with core bench marks	A	1
	2.16	Point A-T-2 and A-B-4 cameras for T2 alignment and leveling	A	-
	2.17	A-T manipulate T2 to alignment	M	2
	2.18	Lock C3 (pyro)	A	-
	2.19	Lock C4 (pyro)	A	-
T2 Installed - Task Time			23 minutes	

Table IIE-2 (Continued)

Primary Function	Secondary Function	Mode	Task Time (Minutes)
3.0 Install Beam T_x3	3.1 Position A-T shoulder at P-1	A	1
	3.2 Position A-T end effector at B-T	A	2
	3.3 Extract T_x3	M	2
	3.4 Open T_x3	M	1
	3.5 Lock opened T_x3 (pyro)	A	-
	3.6 Separate T_x3 from B-T	M	1
	3.7 Position T_x3 at ends of T1 and T2	A	3
	3.8 Place T_x3 into locator position at T1 (Locator pins at opposite ends at alternate cubes)	M	1
	3.9 Weld T_x3 to T1	A	-
	3.10 Point A-T-1 camera for T_x3 alignment	A	-
	3.11 Position A-T shoulder at P-2	A	1
	3.12 Position A-T end effector at T_x3	A	1
	3.13 A-T manipulate T_x3 to alignment	M	2
	3.14 Weld T_x3 to T2	A	-

T_x3 Installed - Task Time 16 minutes
Total Accumulated Time 1 hour 02 minutes

Primary Function	Secondary Function	Mode	Task Time (Minutes)
4.0 Install Beams R1 and R2	- Following steps 4.1 through 4.12 are simultaneously performed by A-B to prepare R2 for installation -		
	4.1 Position A-T shoulder at P-1	A	1
	4.2 Position A-T end effector at B-T	A	} 2
	4.3 Rotate B-T to expose proper quadrant	A	
	4.4 Extract R1	M	1
	4.5 Open R1	M	1
	4.6 Lock opened R1 (pyro)	A	-
	4.7 Extract C5 and attach to R1	M	3
	4.8 Extract C2A (for next cube) and attach to R1	M	3
	4.9 Extract C8A and attach to R1	M	3
	4.10 Extract C6 and attach to R1	M	3
	4.11 Separate R1 from B-T	M	1
	4.12 Position R1 at end of T_x3 (R2 is now at T2)	A	3
	4.13 Hold R1 to T_x3 at weld pads	M	1
	4.14 Weld R1 to T_x3	A	-
	4.15 Position A-T shoulder at P-2	A	1
	4.16 Position A-T end effector at R2	A	2
	4.17 A-T grasps R2 and holds R2 to T_x3 pads (A-B releases R2)	M	1
	4.18 Weld R2 to T_x3	A	-
	4.19 A-T extend C7 to opposite corner	M	2
	4.20 Weld C7 end	A	-
	4.21 A-T extend C8 to opposite corner	M	3
	4.22 Weld C8 end	A	-
	4.23 Point A-B-4 and A-B-3 cameras for R2 alignment	A	-
	4.24 A-B manipulate R2 to alignment	M	2
	4.25 Lock C7 (pyro)	A	-
	4.26 Lock C8 (pyro)	A	-
	4.27 Position A-T shoulder at P-1	A	} 1
	4.28 Position A-B shoulder at P-3	A	

Table IIE-2 (Continued)

Primary Function		Secondary Function	Mode	Task Time (Minutes)
	4.29	A-T extend C5 to opposite corner	M	2
	4.30	Weld C5 end	A	-
	4.31	Extend C6 to opposite corner	M	4
	4.32	Weld C6 end	A	-
	4.33	Point A-B-3 and A-B-4 cameras for R1 alignment	A	-
	4.34	A-B manipulate R1 to alignment	M	2
	4.35	Lock C5 (pyro)	A	-
	4.36	Lock C6 (pyro)	A	-
		R1 and R2 Installed - Task Time 42 minutes; Total Accumulated Time 1 hour 44 minutes		
5.0 Install Beam T_x4	5.1	Position A-B shoulder at P-4	A	1
	5.2	Position A-B end effector at B-B	A	2
	5.3	Extract T_x4	M	2
	5.4	Open T_x4	M	1
	5.5	Lock opened T_x4 (pyro)	A	-
	5.6	Separate T_x4 from B-B	M	1
	5.7	Position T_x4 at ends of R1 and R2	A	3
	5.8	Place T_x4 into locator position at R2	M	1
	5.9	Weld T_x4 to R2	A	-
	5.10	Position A-B shoulder at P-3	A	1
	5.11	Point A-T-1 and A-B-4 cameras for T_x4 alignment	A	-
	5.12	A-B manipulate T_x4 to alignment	M	3
	5.13	Weld T_x4 to R1	A	-
		T_x4 Installed - Task Time 15 minutes; Total Accumulated Time 1 hour 59 minutes		
6.0 Install Beam T5	6.1	Position A-B shoulder at P-4	A	1
	6.2	Position A-B end effector at B-B	A	2
	6.3	Extract T5	M	2
	6.4	Open T5	M	1
	6.5	Lock opened T5 (pyro)	A	-
	6.6	Extract C9 and attach to T5	M	3
	6.7	Extract C10A and attach to T5	M	3
	6.8	Separate T5 from B-B	M	1
	6.9	Position T5 at P-3	A	3
	6.10	Insert T5 into P-3	M	1
	6.11	Fasten T5 in place	A	-
	6.12	Extend C9 to opposite corner	M	3
	6.13	Weld C9 end	A	-
	6.14	Point A-B-3 to align T5	A	1
	6.15	Point A-T-1 to align T1	A	1
	6.16	A-T manipulate T1 to level	M	2
	6.17	A-B manipulate T5 to alignment	M	2
	6.18	Weld T5 to T_x4	A	-
	6.19	Lock C9 (pyro)	A	-
		T5 Installed - Task Time 26 minutes; Total Accumulated Time 2 hours 25 minutes		

Table IIE-2 (Concluded)

Primary F[illegible]ion		Secondary Function	Mode	Task Time (Minutes)
7.0 Install Beam T6	7.1	Position A-T shoulder at P-2	A	
	7.2	Position A-B shoulder at P-4	A	1
	7.3	Position A-B end effector at B-B	A	2
	7.4	Extract T6	M	2
	7.5	Open T6	M	1
	7.6	Lock opened T6 (pyro)	A	-
	7.7	Extract C10 and attach to T6	M	3
	7.8	Extract C9A and attach to T6	M	3
	7.9	Separate T6 from B-B	M	1
	7.10	Position T6 at P-4	A	3
	7.11	Insert T6 into P-4	M	1
	7.12	Fasten T6 in place	A	-
	7.13	Extend C10 to opposite corner	M	3
	7.14	Weld C10 end	A	-
	7.15	Point A-B-4 to align T6	A	1
	7.16	Point A-T-2 to align T2	A	1
	7.17	A-T manipulate T2 to level	M	2
	7.18	A-B manipulate T5 to alignment	M	2
	7.19	Weld T6 to T_x4	A	-
	7.20	Lock C10 (pyro)	A	-

T6 Installed - Task Time 26 minutes

Total Accumulated Time 2 hours 51 minutes

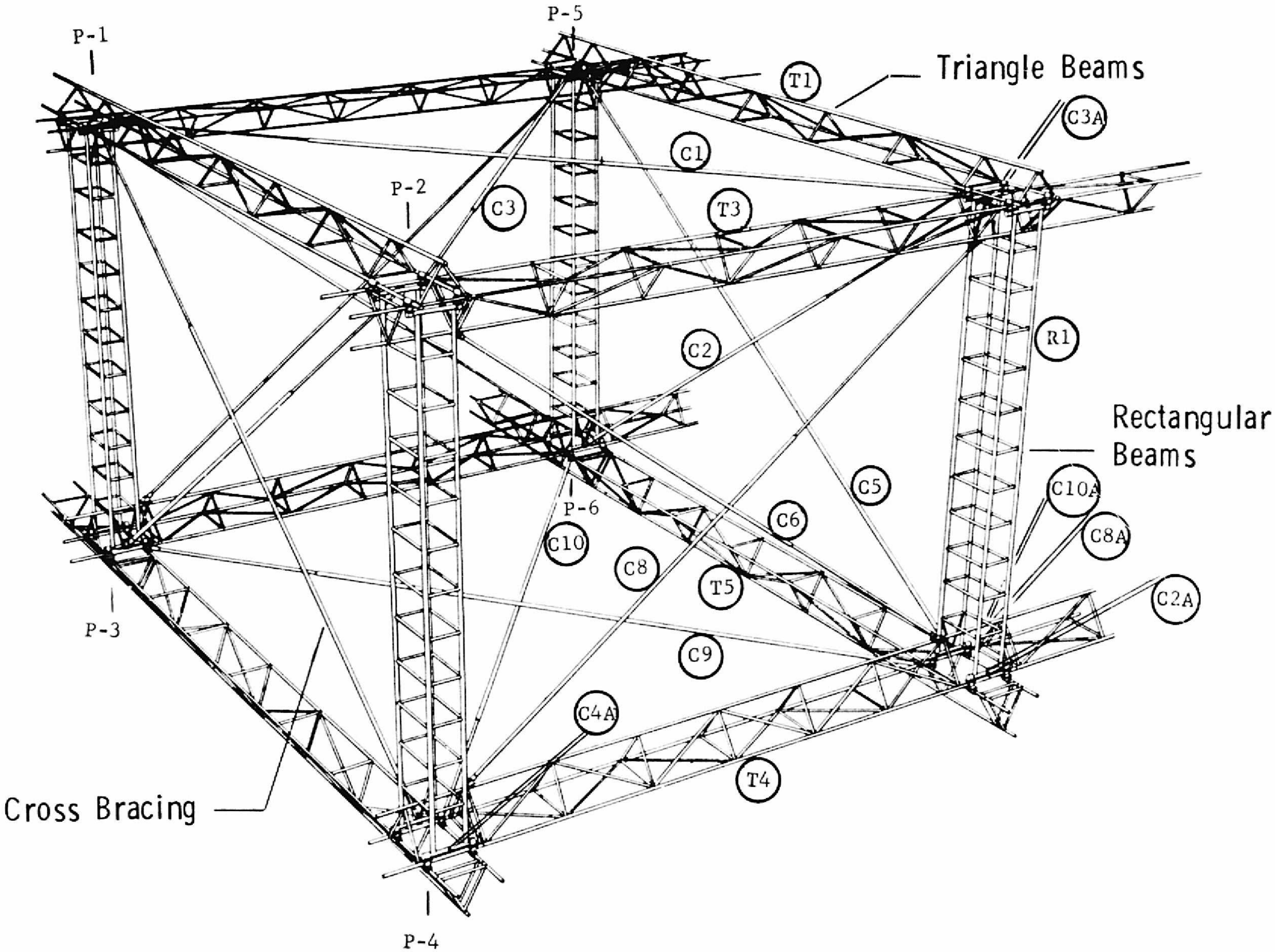


Figure IIE-4 Partial-Cube Structure

Table IIE-3 Partial-Cube Assembly Procedures

Primary Function		Secondary Function	Mode	Task Time (Minutes)
1.0 Install Beam T1	1.1	Position A-T shoulder at P-1	A	1
	1.2	Position A-T end effector at B-T	A	
	1.3	Rotate B-T to expose proper quadrant	A	2
	1.4	Extract T1	M	2
	1.5	Open T1	M	1
	1.6	Lock opened T1 (pyro)	A	-
	1.7	Extract C1 and attach to T1	M	3
	1.8	Extract C3A (for next cube) and attach to T1	M	3
	1.9	Separate T1 from B-T	M	1
	1.10	Position T1 at P-5	A	2
	1.11	Insert T1 into P-5	M	1
	1.12	Fasten T1 in place	A	-
	1.13	Extend C2 to opposite corner (shoulder travel) (installed with original R1)	M	3
	1.14	Weld C2 end	A	-
	1.15	Position A-T shoulder at P-1	A	1
	1.16	Extend C3 to opposite corner (installed with original T2)	M	2
	1.17	Weld C3 end and lock C3 (pyro)	A	-
	1.18	Extend C1 to opposite corner	M	2
	1.19	Weld C1 end	A	-
	1.20	A-T-1 camera leveled with core bench marks	A	1
	1.21	Point A-T-1 and A-T-2 cameras for T1 alignment	A	-
	1.22	A-T manipulate T1 to alignment (shoulder to P-2)	M	3
	1.23	Lock C2 (pyro)	A	-
	1.24	Lock C1 (pyro)	A	-
		T1 Installed - Task Time 28 minutes		
2.0 Install Beam T3	2.1	Position A-T shoulder at P-1	A	1
	2.2	Position A-T end effector at B-T	A	2
	2.3	Extract T3	M	2
	2.4	Open T3	M	1
	2.5	Lock opened T3 (pyro)	A	-
	2.6	Separate T3 from BT	M	1
	2.7	Position A-T shoulder at P-2	A	1
	2.8	Position T3 at P-2	A	2
	2.9	Insert T3 into P-2	M	1
	2.10	Fasten T3 in place	A	-
	2.11	Extend C8 to opposite corner (C8 initially attached)	M	2
	2.12	Weld C8 end	A	-
	2.13	Point A-T-1 and A-T-2 cameras for T3 alignment	A	-
	2.14	Manipulate T3 to alignment	M	5
	2.15	Weld T3 to T1	A	-
	2.16	Lock C8 (pyro)	A	-
		T3 Installed - Task Time 18 minutes		
		Total Accumulated Time 46 minutes		

Table IIE-3 (Continued)

Primary Function		Secondary Function	Mode	Task Time (Minutes)
3.0 Install Beam R1	3.1	Position A-T shoulder at P-1	A	1
	3.2	Position A-T end effector at B-T	A	2
	3.3	Extract R1	M	1
	3.4	Open R1	M	1
	3.5	Lock opened R1 (pyro)	A	-
	3.6	Extract C5 and attach to R1	M	3
	3.7	Extract C2A and attach to R1	M	3
	3.8	Extract C8A and attach to R1	M	3
	3.9	Extract C6 and attach to R1	M	3
	3.10	Separate R1 from B-T	M	1
	3.11	Position R1 at end of T3 (shoulder to P-2)	A	4
	3.12	Hold R1 to T3 at weld pads	M	1
	3.13	Weld R1 to T3	A	-
	3.14	Extend C6 to opposite corner	M	2
	3.15	Weld C6 end	A	-
	3.16	Move A-T carriage to span P-1 to P-5	A	5
	3.17	Extend C5 to opposite corner	M	2
	3.18	Weld C5 end	A	-
	R1 Installed -	Task Time 32 minutes		
		Total Accumulated Time 1 hour 18 minutes		
4.0 Install Beam T4	4.1	Position A-B end effector at B-B	A	2
	4.2	Extract T4	M	2
	4.3	Open T4	M	1
	4.4	Lock opened T4 (pyro)	A	-
	4.5	Separate T4 from B-B	M	1
	4.6	Position T4 at P-4	A	2
	4.7	Insert T4 into P-4	M	3
	4.8	Fasten T4 in place	A	-
	4.9	Hold T4 to R1 at weld pads	M	1
	4.10	Weld T4 to R1	A	-
	4.11	A-B-3 camera leveled with core bench marks	A	1
	4.12	Point A-B-3 and A-B-4 cameras for T4 alignment	A	-
	4.13	Manipulate T4 to alignment	M	5
	4.14	Lock C6 (pyro)	A	-
	4.15	Lock C5 (pyro)	A	-
	T4 Installed -	Task Time 18 minutes		
		Total Accumulated Time 1 hour 36 minutes		
5.0 Install Beam T5	5.1	Position A-B end effector at B-B	A	2
	5.2	Extract T5	M	2
	5.3	Open T5	M	1
	5.4	Lock opened T5 (pyro)	A	-
	5.5	Extract C10A and attach to T5	M	3
	5.6	Extract C9 and attach to T5	M	3
	5.7	Separate T5 from B-B	M	1
	5.8	Position T5 at P-6	A	2
	5.9	Insert T5 into P-6	M	1
	5.10	Fasten T5 in place	A	-
	5.11	Position A-B shoulder at P-3	A	1

Table IIE-3 (Concluded)

Primary Function	Secondary Function	Mode	Task Time (Minutes)
	5.12 Extend C10 to opposite corner	M	2
	5.13 Weld C10 end	A	-
	5.14 Lock C10 (pyro)	A	-
	5.15 Extend C9 to opposite corner	M	2
	5.16 Weld C9 end	A	-
	5.17 Point A-B-3 and A-B-4 cameras for T5 alignment	A	-
	5.18 A-B manipulate T5 for alignment	M	5
	5.19 Weld T5 to T4	A	-
	5.20 Lock C9 (pyro)(T5 installed)	A	-
	5.21 Cameras A-T-2 and A-B-4 read angles to opposite corners for error inputs into computer base	A	-
	5.22 Move assemblers and beam pallets to next cube	A	10
T5 Installed - Task Time	35 minutes		
Total Accumulated Time	2 hours 11 minutes		

F. SIMULATION OF ASSEMBLY IN ORBIT (TASK 7)

1. Introduction

A series of simulations has been conducted in accordance with the Simulation Plan. The intent of the contract is to study and derive designs and methods to (1) assemble very large space structures, and (2) maintain geosynchronous satellites. Several approaches to both assembly and maintenance tasks have been developed as part of the analysis and design tasks. Analysis of the requirements for simulation in maintenance showed that most major tasks have been previously simulated. Due to the magnitude of the assembly tasks and the fact that these structures are planned well into the future, there is little existing related data from which to draw. As our assembly approaches have evolved, anticipated problem areas have come to light. These anticipated problem areas have formed the basis for simulation tasks.

Our primary concern is related to remote handling of large, 60-ft long, beams in space. This handling includes extraction from a stowage area, translation and alignment, and attachment on one or both ends. We know of no simulations which have addressed this activity in total. The main objective of the assembly simulation is to determine whether or not the beam handling tasks can be accomplished while utilizing the proposed equipments and techniques, to develop recommendations for manipulator design, alignment aid design, and to determine further simulations.

Compounding the actual assembly problem is the fact that the support equipment or 'tools' have not yet been developed. This creates a situation where we are planning the use of support hardware e.g., a mobile assembler (MA), Shuttle Remote Manipulator System (RMS), from which our performance requirements may, in fact, create driving design requirements for this support equipment.

We have included in this chapter both the Radio Astronomy Telescope and Microwave Antenna simulations.

2. Objectives

The primary simulation objective was to demonstrate the usefulness of (1) the Shuttle Remote Manipulator System (RMS) for assembly of the Radio Astronomy Telescope (RAT) core, and (2) the Mobile Assembler (MA) for handling of long beams in assembly of the Microwave Power Transmission System (MPTS). Secondary objectives were (1) to demonstrate the feasibility of beam and attachment designs for space assembly, (2) to evaluate support hardware and task parameters such as TV camera locations and configurations, timelines, etc., (3) to evaluate the MA control system, (4) to determine the operator's displayed information requirements, (5) to determine the requirements for supplemental alignment aids at the attachment interface, and (6) to develop preliminary requirements for manipulator design parameters such as maximum joint angle positions and rates.

The resulting data showed all tasks to be not only feasible but fairly easy for the trained operator to perform within the constraints of the simulation. The secondary objectives were met as well. Details will follow in this report.

3. Description and Operation of the Simulation Facility

The simulation facility consists of a Slave Manipulator Arm, a Test Conductor's Control Console, an Operator's Console, Video and Audio Communications System and Analog Computers located in four adjoining rooms as shown in Figure IIF-1. An information flow block diagram identifying the signals going to and from each piece of hardware is shown in Figure IIF-2. This section discusses each of these items of equipment.

a. Slave Manipulator Arm - Martin Marietta has designed and fabricated a 12-ft counterbalanced Slave Manipulator Arm (SMA), to be used for resolving the questions of operational applications, capabilities and limitations for such remote manned systems as the Shuttle RMS, the EOTS, the Advanced Space Tug and Planetary Rovers. As a developmental tool for the Shuttle RMS, the SMA represents an approximate one-quarter scale working model for simulating and demonstrating payload handling, docking assistance, satellite assembly, and servicing.

The design of the SMA was based on concepts developed for a 40-ft NASA technology arm to be used for zero-g Shuttle manipulator simulations.

1) Description of arm and counterbalance system - The SMA uses an articulated counterbalance scheme for shoulder and elbow and a self-counterbalanced design for the wrist. The articulated counterbalance scheme is essentially a second arm at the end of the shoulder extension with one or two counterbalance weights which are driven (via mechanical linkage) in phase with the lower arm (elbow to wrist). This system provides an arm whose shoulder and elbow torques need not, in any orientation, overcome the force of gravity on either the upper or lower arm, and whose motion is completely unrestricted. The SMA with its articulated counterbalance is shown in Figure IIF-3. A moment diagram for the arm showing the relationship between the main portion of the arm and the counterbalance is given by Figure IIF-4.

The SMA counterbalance linkage configuration is a three-bar direct-drive system that was selected to meet the design requirements for low friction and high stiffness. Each of the three bars is mounted on a separate crank plate. These assemblies were put together to form a crankshaft-rod type system. The pin locations in the plates are 120 degrees apart. Figure IIF-5 is a schematic of the linkage system.

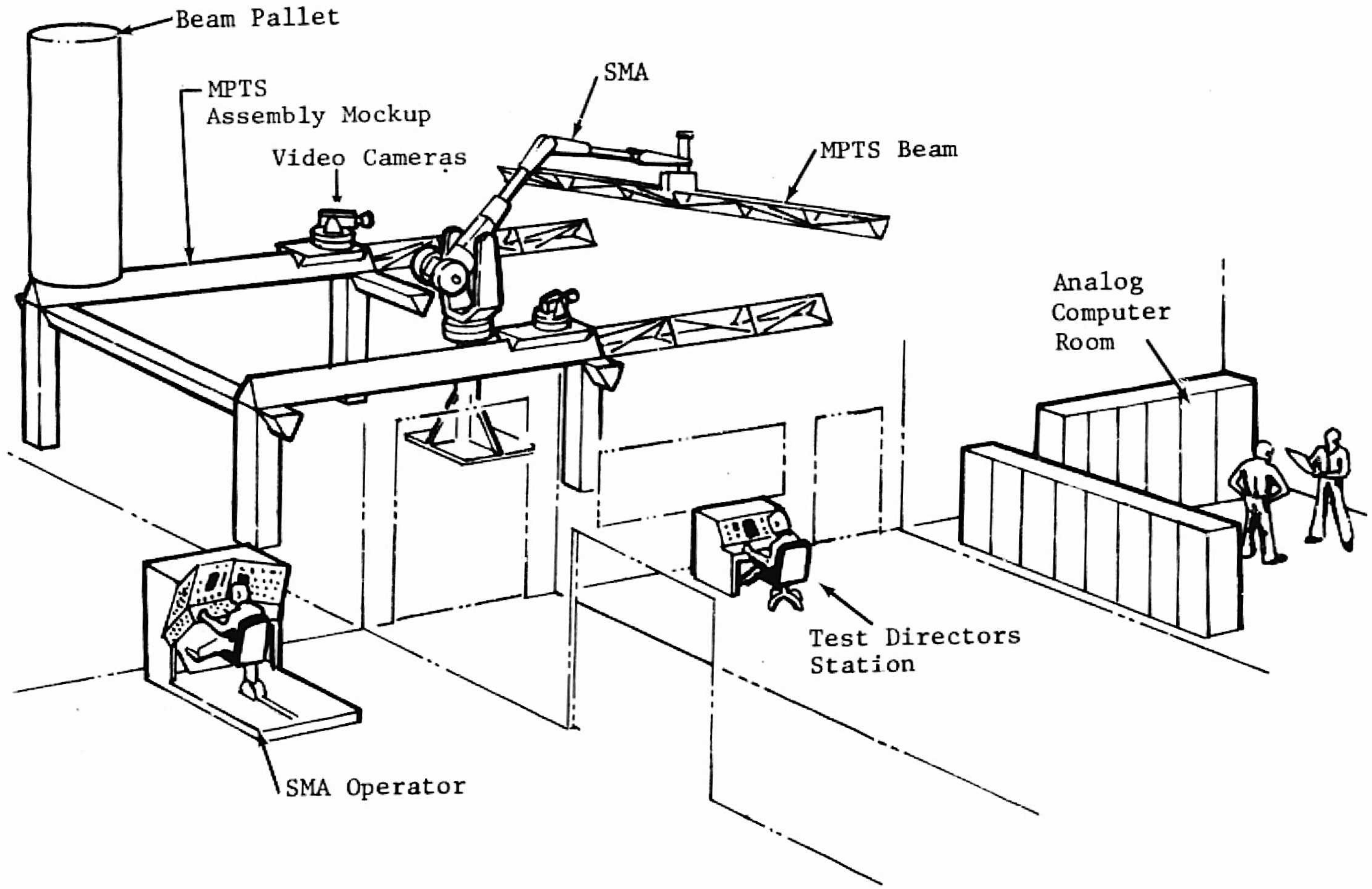
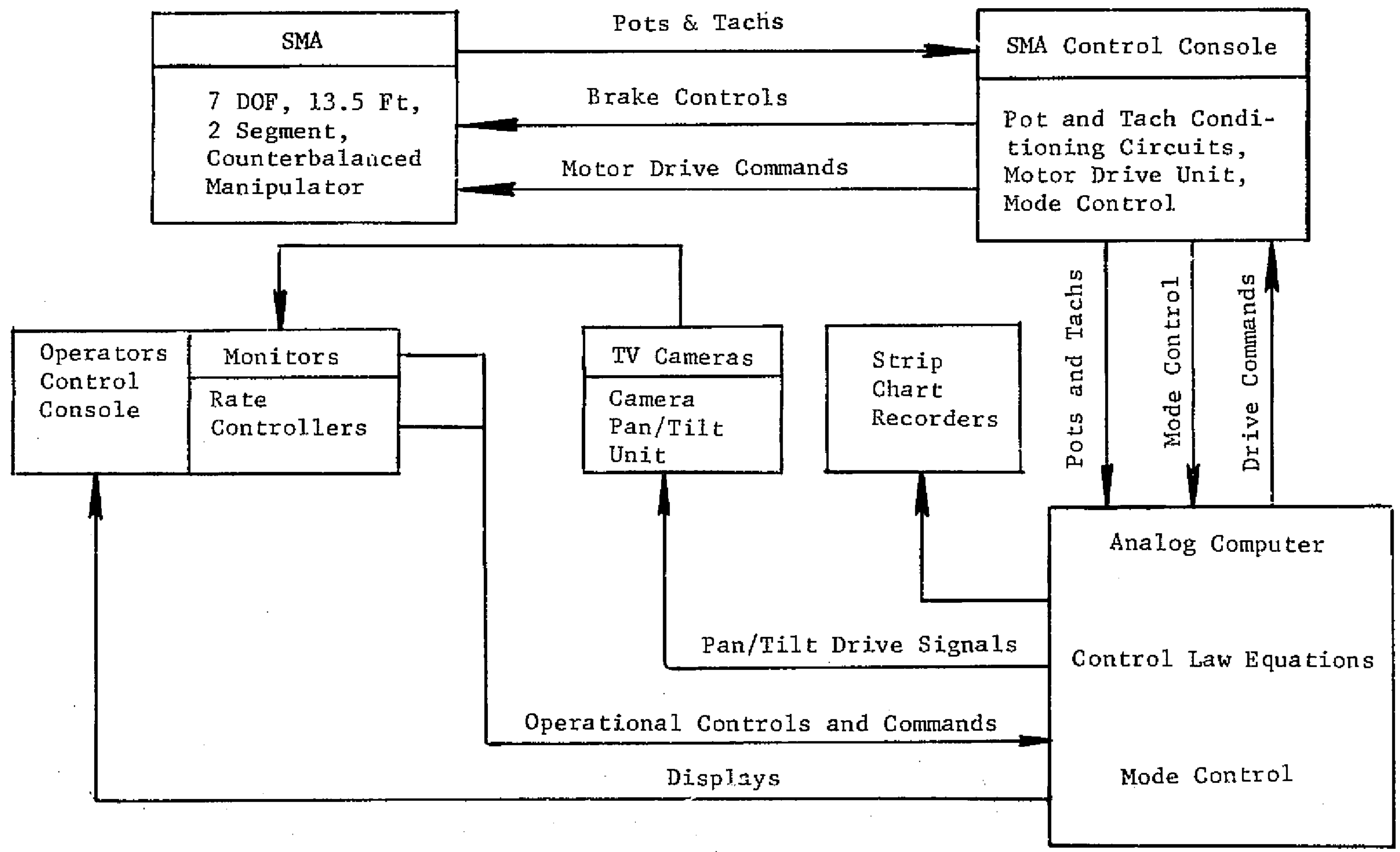


Figure IIF-1 Simulation Facility



SMA = Slave Manipulator Arm
SCN = Servo Compensating Networks

Figure IIF-2 Simulation Hardware Components and Information Flow

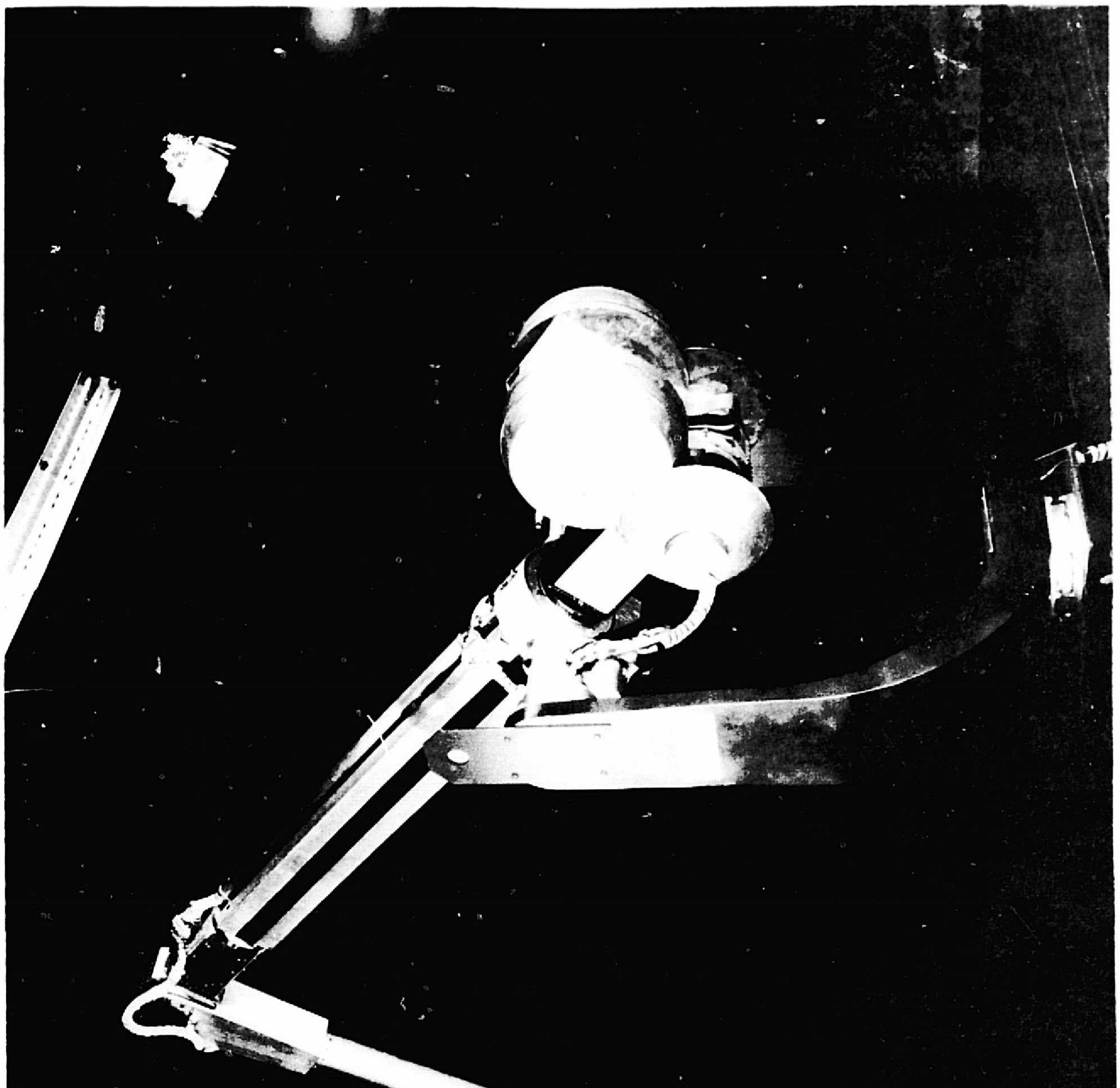


Figure IIF-3 Slave Manipulator Arm

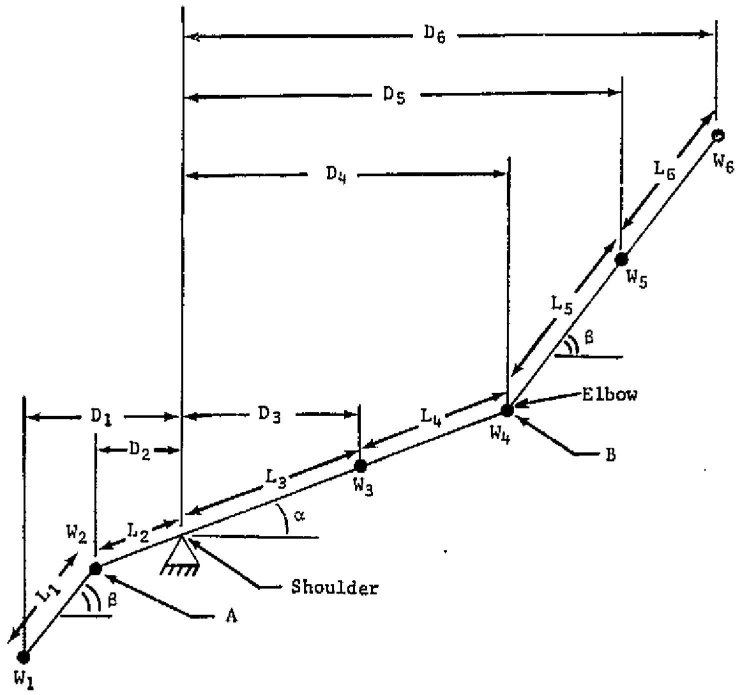


Figure IIF-4 Articulated Counterbalance Model

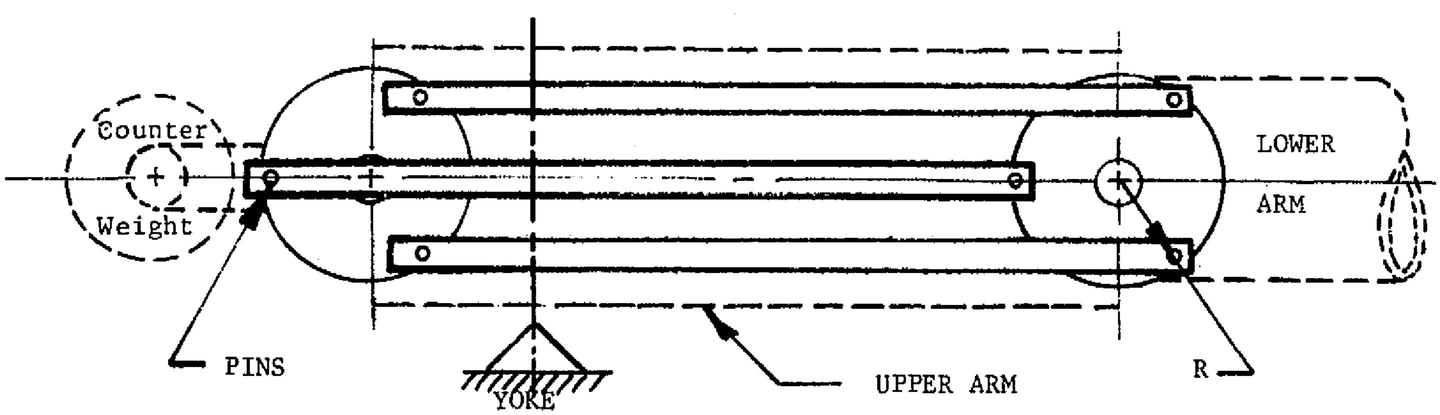


Figure IIF-5 Counterbalance Linkage Concept

Table IIF-1 illustrates the general joint capabilities. The drives are the most significant portion of the design because of their close approximation to flight hardware. All joints contain 60 vdc motors, potentiometers, tachometer-generators and fail safe 28 vdc friction brakes. The gear ratios are in the range of 115:1. Each joint is servo-controlled, and may accept commands from manual or computer sources. Of particular interest in the drives during simulations are joint flexibility/stiffness, gear backlash/backdrive, finite motor torque, friction/stiction and response.

The wrist consists of 3 degrees of freedom. Counterbalancing is accomplished by the proper placement of equipment around each axis, as shown in Figure IIF-6. By placing the pitch axis at the center of gravity of the roll drive and the end effector/TV camera assembly, the pitch axis is balanced. By locating the yaw axis at the center of gravity of the pitch motion equipment, and the pitch/yaw drive assembly, the yaw axis is balanced. The pitch and yaw drives are separated from their respective drive centers by the use of steel drive tapes. All equipment is mounted on the main wrist support member which pivots around the yaw axis. Figure IIF-7 is a photograph of the SMA wrist and end effector.

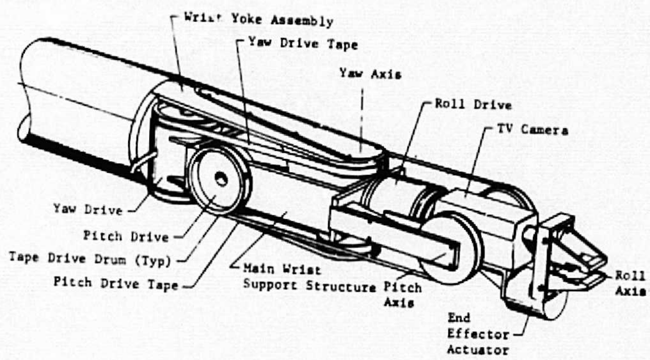


Figure IIF-6 Self-Counterbalanced Wrist Assembly

2) <u>SMA Characteristics</u> - The SMA static deflection data is presented in Table IIF-2. The table shows the deflection due to both structural and gear train flexibility. The natural frequency of the arm (when fully extended and brakes engaged) is approximately 0.7 Hz with a critical damping factor of 15%.

The motion resolution of the SMA was determined by measuring the minimum possible movement of the thermal device for very small input commands. It was found that all control systems could input commands smaller than that required to move the arm. Eventually, the small commands (inputted by small pulses) increased the joint torque until stiction was overcome and the arm

Table IIF-1 Electromechanical Joint Capabilities

	SHOULDER		ELBOW		WRIST	
	YAW ψ_s	PITCH θ_s	PITCH θ_E	ROLL ϕ_W	PITCH θ_W	YAW ψ_W
Stall Torque (Ft-lb)	110	110	66	33	33	15
Gear Ratio	110	110	112.7	112.7	112.7	112.7
Angular Travel (Degrees)	±200	+ 75 -150	+ 10 -160	±130	±80	±200
Maximum Angular Rate (Deg/Sec)	30	30	30	30	30	30
Joint Backlash (Arc Min)	3	2	2	0	2	1
Backdrive Torque Brake Off (Ft-lb)	18	2.5	3.5	1.07	0.9	1.5
Backdrive Torque Brake On (Ft-lb)	69	60	75	40	42	40

Table IIF-2 SMA Static Deflection

SEGMENTS DEFLECTED	DEFLECTION (IN/LB) (TOTAL WITH GEAR TRAINS)
Shoulder to Wrist	.130
Shoulder to Elbow	.015
Elbow to Wrist	.045

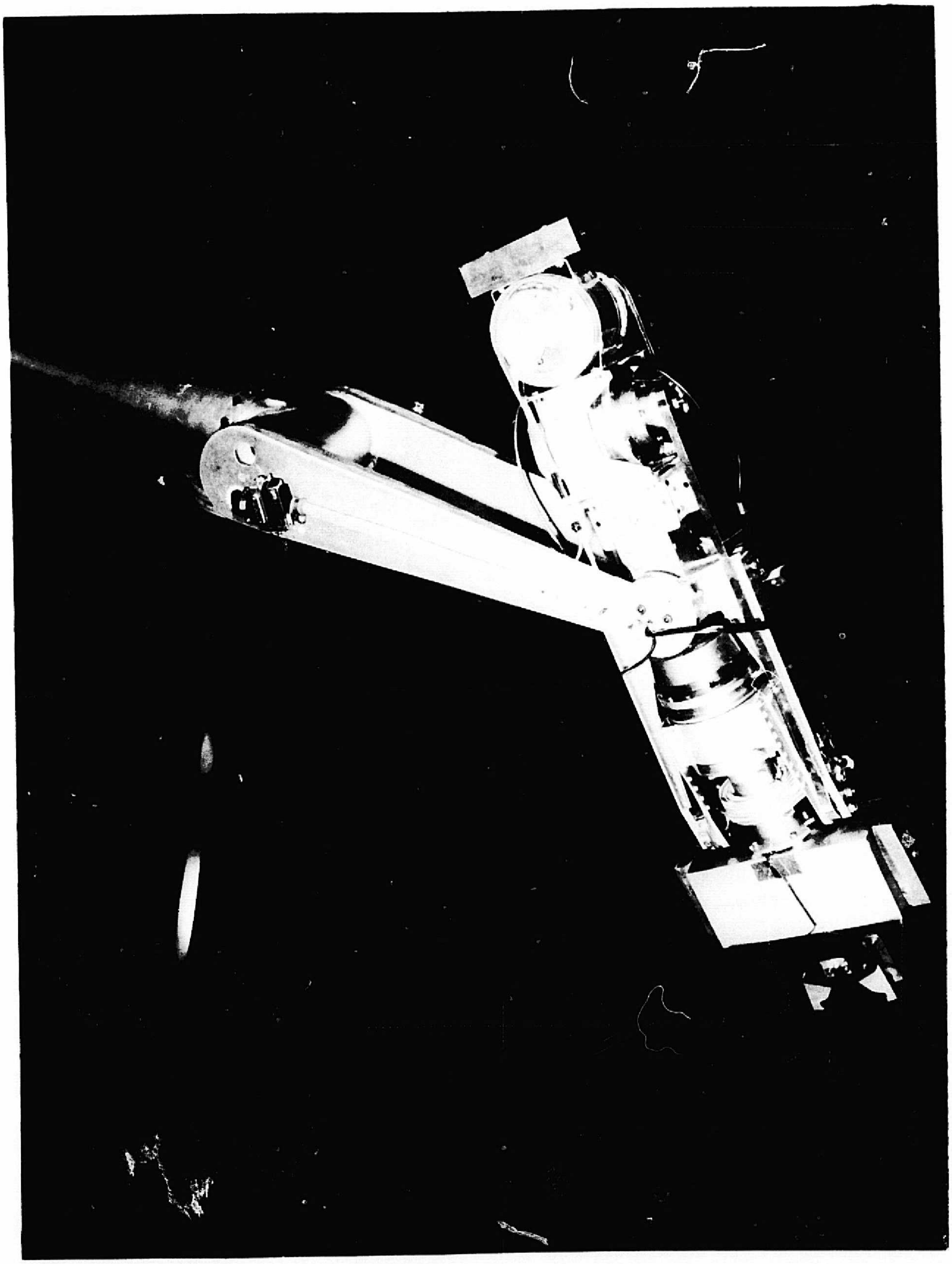


Figure IIF-7 SMA Wrist

moved. This resulting motion for all six degrees of freedom is listed in Table IIF-3. The SMA static force resolution was determined in the same manner, i.e., by small pulse input commands, when the arm was rigidly attached to a load cell array. In this case, the minimum force change from an impulse was better than the resolution of the load cell array. It is estimated that SMA forces at the terminal device can be controlled to less than 0.2 lbs and the torques to less than 0.4 ft-lbs.

Table IIF-3 SMA Motion Resolution

Range	- 3/16 inch	Wrist Pitch	0.1 degrees
Azimuth	- 1/16 inch	Wrist Yaw	0.2 degrees
Elevation	- 1/8 inch	Wrist Roll	0.5 degrees

b. Test Conductor's Control Console - The Test Conductor's Control Console (TCCC) provides the equipment necessary to (1) power up the SMA, (2) select operating modes, (3) monitor system operation and provide limit warnings, (4) allow manual SMA control, (5) bring the SMA to a safe stop in an emergency, (6) control the associated analog computer, and (7) house the system electronics. Figure IIF-8 is a photograph of the TCCC.

The TCCC was designed to be as flexible as possible to accommodate experimental configuration changes and operational improvements. Plug-in circuit boards are used extensively and all cabling is terminated in connectors so that large sections (such as an entire panel) can be completely removed for modification or maintenance. System reliability is enhanced by the use of solid state switching in all signal circuits.

Power for the SMA and TCCC is obtained from 117 vac and 28 vdc mains. The internal power supplies provide dc voltages of +60, +15, -15, +10, -10, and +5.

The TCCC makes provision for several modes of operation including both rate and position servo control. In the normal operating situation with an analog computer in the loop, the command signals (either rate or position) from controllers located in the Operator's Station (OS) are routed to the computer. The computer represents a servo rate command which is amplified in the TCCC and applied to the proper joint servomotor. As an alternative, the computer may utilize the joint position potentiometer outputs to produce a servo signal based on arm position.

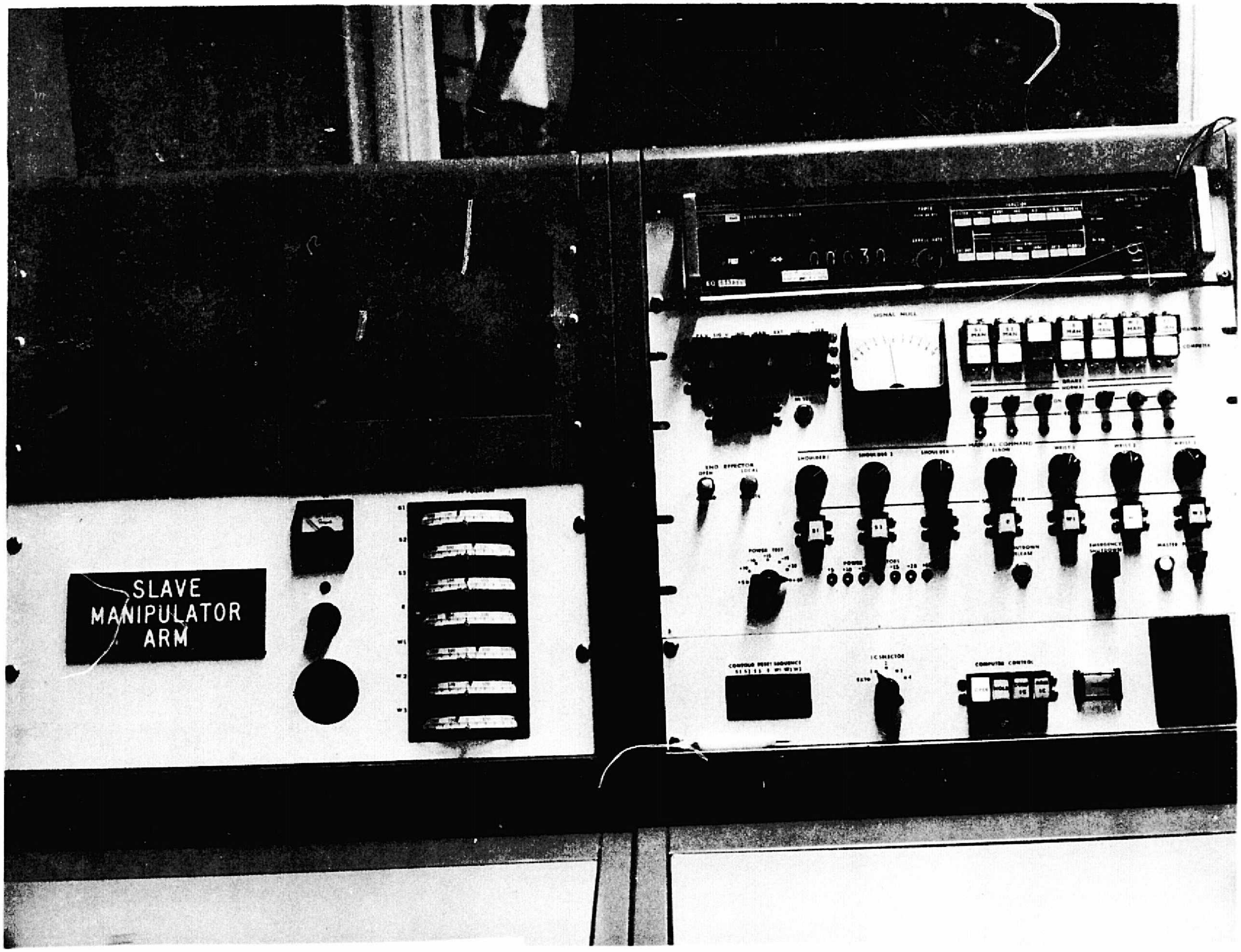


Figure IIF-8   Test Conductor's Control Console

There is also a MANUAL mode, in which mode position commands come from 3-turn potentiometers on the TCCC Main Control Panel. SMA joints may be placed in the MANUAL mode individually while other joints remain under computer control.

Other provisions are included for system safety. A power interrupt circuit removes servomotor power and applies the electromagnetic joint brakes when activated. Power interrupt can be initiated manually by observers located in various parts of the facility or automatically when a joint angle limit occurs. When a joint angle is approaching its limit, an early warning is provided by the limit circuitry. An audible beeping sound occurs and a red warning light flashes to indicate an approaching limit. At this time, the operator can reverse the action and drive normally back to a safe condition at which time the warnings cease. If the limit is exceeded, however, the power interrupt will be initiated as described above. It is then necessary to bring the SMA out of the limit under manual control.

c. Operator's Console - This control station was designed and laid out for optimum manned interface characteristics, such as controller reach and visual angle limits for our mono and stereo TV monitors. The console panels are removable, which allows various control and display layouts to be evaluated. The present configuration, which will be used for this in-space assembly simulation, is laid out around the two video monitors shown in Figure IIF-9. These monitors are the operator's only visual feedback since there can be no direct vision in this task. The visual displays located on the center console are as follows:

1) Two 10-inch video monitors;
2) Three displays for SMA tip applied force data, $\pm$ X Y Z axis;
3) Three displays for SMA tip applied moment data, $\pm$ pitch, yaw, roll;
4) A display for terminal device closure force and one for percent closed;
5) Seven displays for SMA joint angle readout;
6) Caution and warning lights including:

    a) Terminal device contact,
    b) SMA commanded velocity, 70% and 100%,
    c) Position controller override.

The controls located on the overall console include:

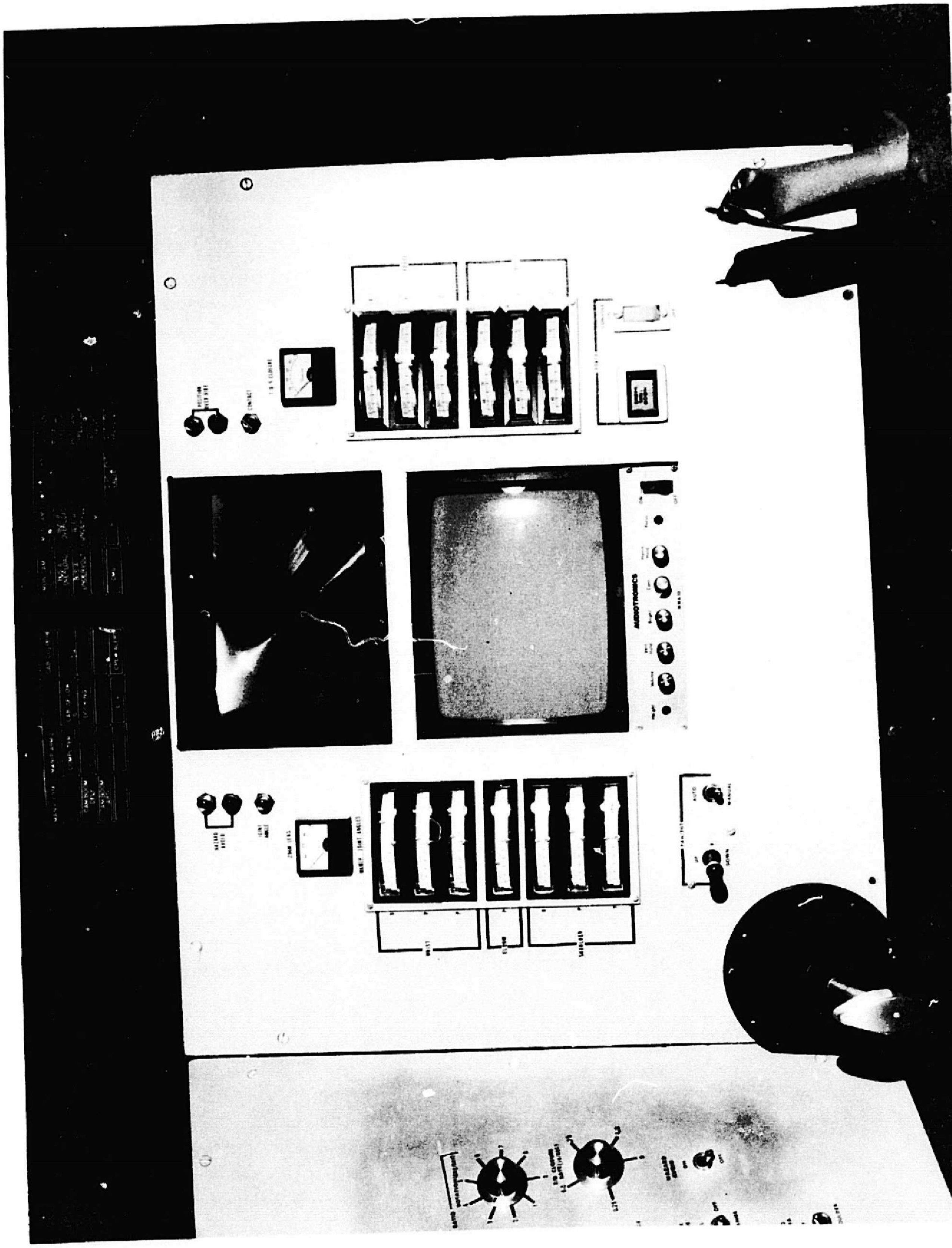


Figure IIF-9 Operator's Console Panel Controls

1) Video camera pan-tilt and lens zoom, iris and focus (manual and auto) for two cameras;
2) Set of Apollo-type rate controllers;
3) Position control ratio select, 1:1 to 1:10;
4) Rate control ratio select, translation (ft/sec) and rotation (deg/sec);
5) Position control force ratio 1:1 to 1:4;
6) Position control torque ratio 1:1 to 1:15;
7) Wrist angular ratio 1:1 to 1:4;
8) Terminal device closure rate 0.25 to 1.0 in./sec;
9) Control mode select (position or rate);
10) Control axis select, tip or shoulder cameras;
11) "Hawk" mode select (full, range, or off);
12) Hazard avoidance, on-off (computer controlled);
13) Video camera select.

d. Video and Audio Communications - The SMA tasks require both audio and video communications for their accomplishment. Direct viewing by the operator cannot be employed because the arm and mockups are scaled down. The SMA itself has provision for two cameras in the vicinity of the wrist and one at the elbow. TV cameras can be placed at other locations on the mockups as needed for a particular simulation. The EAI 525-line, 30 frame-per-second standard applies to all facility cameras and monitors. Two monitors each are located in the TCCC and the operator's console. Cameras are selected by the test subject and controlled either manually by the test subject or by the computers.

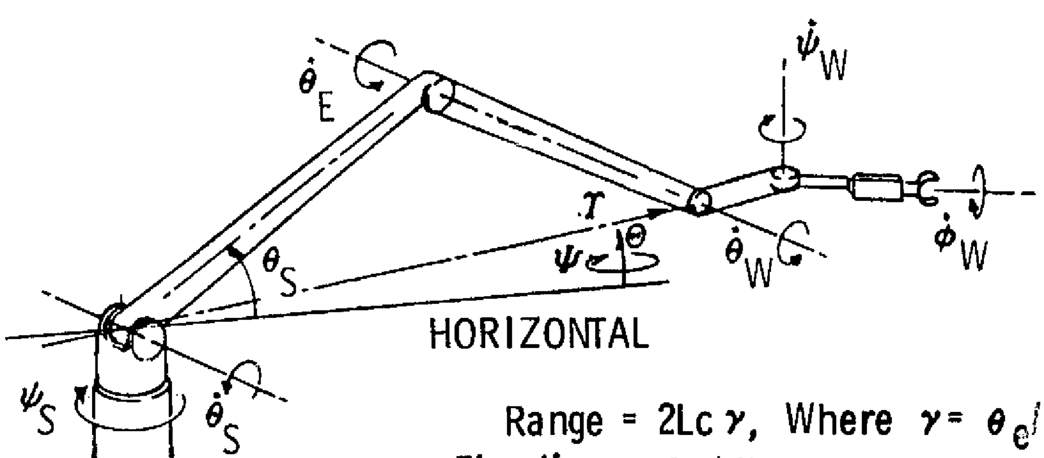
Audio communications are accomplished by a programmable system located in the Control Room. Two-way voice communications will be provided between the test subject, the TCCC operator, the test conductor, the computer room, and other areas as needed during the simulation exercise. The communications system will operate either from headsets with attached boom microphone or from speakers and separate microphones.

e. Analog Computers - Two EAI 231-R analog computers are used in the simulation. The computers are used to program the control law equations, to close control loops around the manipulator joints, and to interface with inputs from the operator's console. The analog program includes approximately 120 summing and inverting amplifiers, 120 potentiometers, 30 multipliers, 20 integrating amplifiers, and 8 resolvers.

4. Control Laws

a. Introduction - Control laws refer to the equations used to interface the control input devices with the manipulator arm gimbal actuators. These laws can range from very simple to quite complex, depending upon the desired versatility to be designed into the manipulator system. Evaluated in a previous simulation were a set of Martin Marietta conceived control equations that are not only comparatively simple but also extremely versatile in that they accommodate both unilateral rate controllers and a bilateral force feedback position controller. The control technique is somewhat unique in that the operator has control of certain selectable variables, an example being the variable rotational and translational rate ratios (or gains) for the rate controller commands. These same control laws were used in the orbital assembly simulation with additional features--such as switchable control axes, to be described later. Until only recently, it was commonly believed that certain space related manipulator tasks, such as module retraction-replacement, beam handling, connector engagement-disengagement, probe insertion, etc., might possibly only be accomplished with a bilateral force reflecting system. A previous Martin Marietta simulation (December, 1974) was performed to compare unilateral rate control and bilateral position control when attempting to perform force related tasks. From the onset of the control law development, it was realized that to provide a fair comparison of the two systems, the unilateral rate controllers must have the dual capability of: 1) commanding manipulator rates when the arm was free to move and, 2) commanding manipulator forces when the objective was to apply a force or torque to a grasped object. For the rate controllers to succeed in performing force related tasks, it was felt that the actual forces and torques generated by the manipulator on its environment must be visually displayed to the operator. To accomplish this, the information must be obtained by actual measurements or computed from related known parameters. Since actual measurements are definitely impractical, the control equations were designed to provide the needed information which was displayed to the operator via three force and three torque meters. The practicality of the rate control approach was demonstrated in the earlier simulation and verified further in the current simulation. The operator's task was further simplified by providing him with optional features such as the attitude hold mode and switchable control axes, to be discussed in a later section.

b. <u>Range-Azimuth-Elevation/Rotation Control Mode</u> - Of many different coordinate systems that can be utilized to control the manipulator arm, the current simulation will be based on a spherical coordinate system which is referred to as Range-Azimuth-Elevation (RAE)/Rotation control. Figure IIF-10 shows the spherical coordinates r, θ, ψ used to define the wrist attachment point; shoulder, elbow, and wrist gimbal angles, and gimbal rates are also depicted in the schematic. In this system translational and rotational motions are separated in that range, azimuth and elevation of the wrist gimbal attachment point provides translational freedom; rotational attitude control is achieved by coupling the input controller on a one-to-one basis with the three wrist angles ψ_w (yaw), θ_w (pitch), and ϕ_w (roll). Auxiliary coordinate axes are used in conjunction with the basic RAE/Rotation control scheme. These coordinate axes are required to input commands from each of several viewing cameras and to, on operator option, facilitate pure translation of the terminal device without incurring undesirable rotations of the terminal device/payload assembly. Both unilateral rate and bilateral (force feedback) position controllers can be used with the RAE/Rotation control technique. Forward, side, and vertical motion of the hand grip correspond to range, azimuth and elevation commands, respectively, for the position controller.



Range = $2Lc\,\gamma$, Where $\gamma = \theta_e/2$

Elevation = $\theta_S + \gamma$

Azimuth = ψ_S

Figure IIF-10 SMA Degrees of Freedom

The simplicity of utilizing spherical coordinates is revealed by the following equations relating gimbal and command degrees of freedom.

$$r = 2L\ (\text{cosine}\ \theta_e/2)$$

$$\theta = \theta_s + \theta_e/2 \qquad \text{IIF-1}$$

$$\psi = \psi_s,$$

where L denotes the length of the (equal-lengthed) upper or lower arm segments.

b. <u>Control System Flow Description</u> - As previously noted, the simulation can be operated using either unilateral rate or bilateral position controllers. Figure IIF-11 illustrates the RAE/rotation control scheme for the unilateral rate control mode. In this mode, commanded translational rates ($\dot{R}$, $\dot{A}$, $\dot{E}$) and rotational rates ($\dot{\psi}_{wc}$, $\dot{\theta}_{wc}$, $\dot{\phi}_{wc}$) are compared with the actual translational and rotational rates of the manipulator arm. A rate error signal is formed and related to manipulator-applied forces and torques. Thus, when in contact with an object, no forces and moments are produced unless the rate controller is deflected and held. The magnitude of the applied forces and moments will then be proportional to controller displacement. Force and moment magnitude data are displaced to the operator. The translational rate commands are related to the gimbal angles and gimbal rates by:

$$\dot{R} = (-\ L \sin \frac{\theta_e}{2})\ \dot{\theta}_e$$

$$\dot{A} = (2L \cos \frac{\theta_e}{2})\ \dot{\psi}_s \qquad \text{IIF-2}$$

$$\dot{E} = (2L \cos \frac{\theta_e}{2})\ (\dot{\theta}_s + \frac{1}{2}\dot{\theta}_e)$$

Alternative to direct $\dot{R}$, $\dot{A}$, $\dot{E}$ commands, the coordinate transformation T (Eq. 5, Figure IIF-11) derives $\dot{R}$, $\dot{A}$, $\dot{E}$ values in the base axis system from cartesian $\underline{X}_c$ commands in the terminal device coordinate system. The T transformation will also be used when commands are initiated in the coordinate systems associated with the two cameras located at the wrist assembly. Additionally, this transformation used in conjunction with the Hawk commands

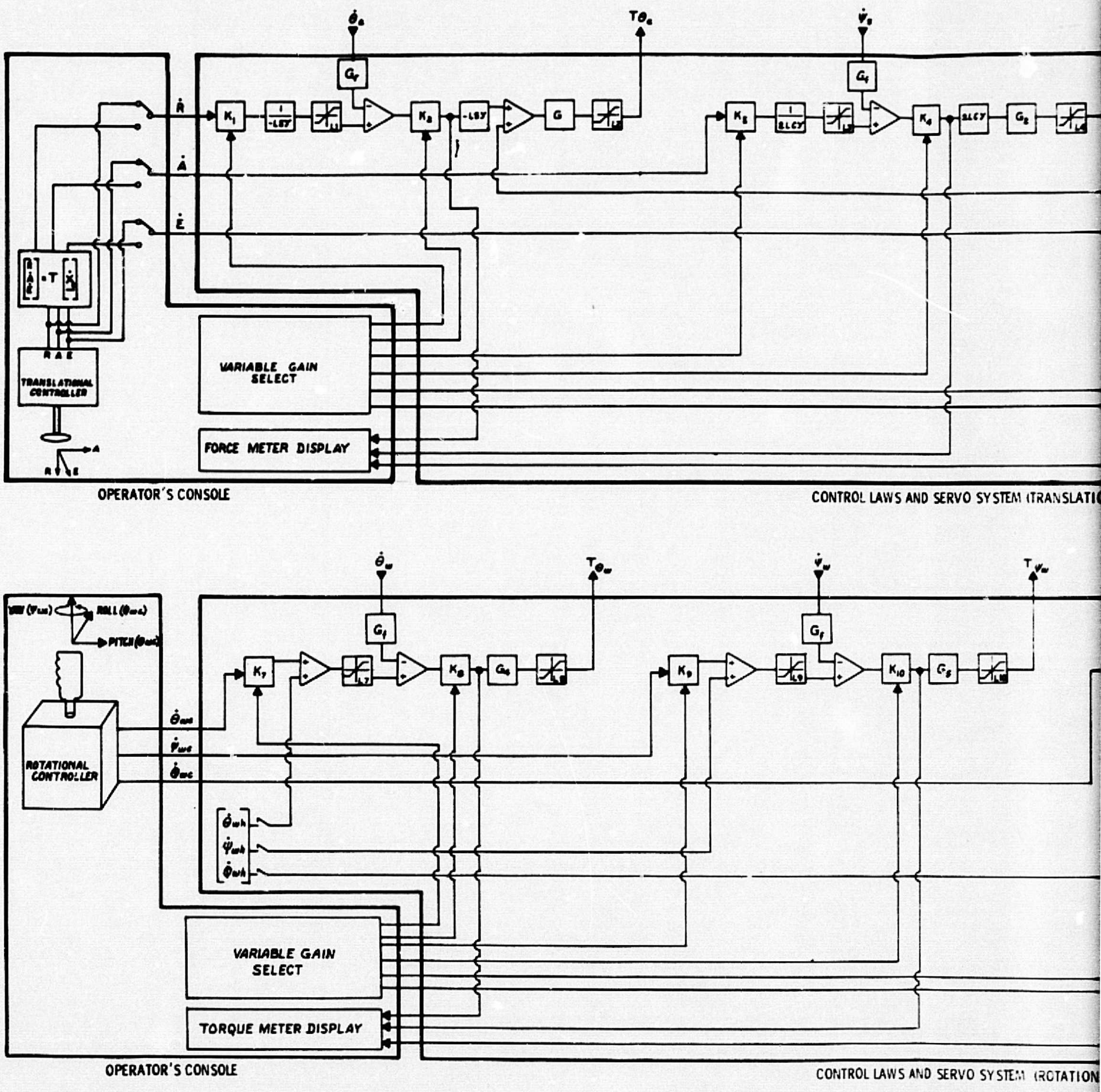


Figure IIF-11 RAE/Rotation Control System

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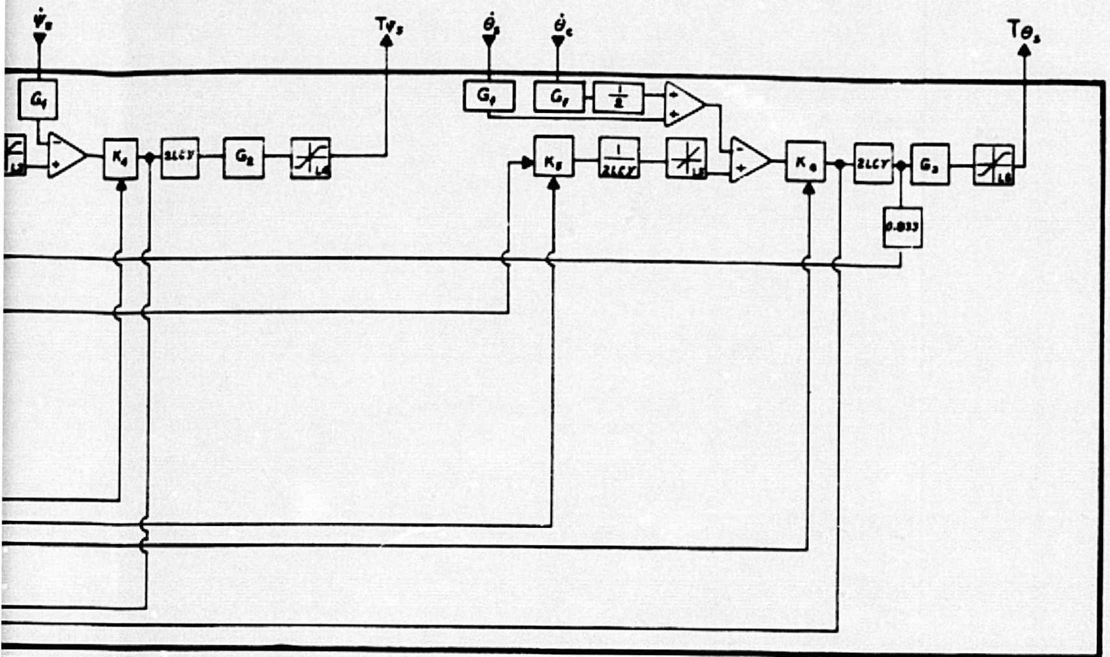
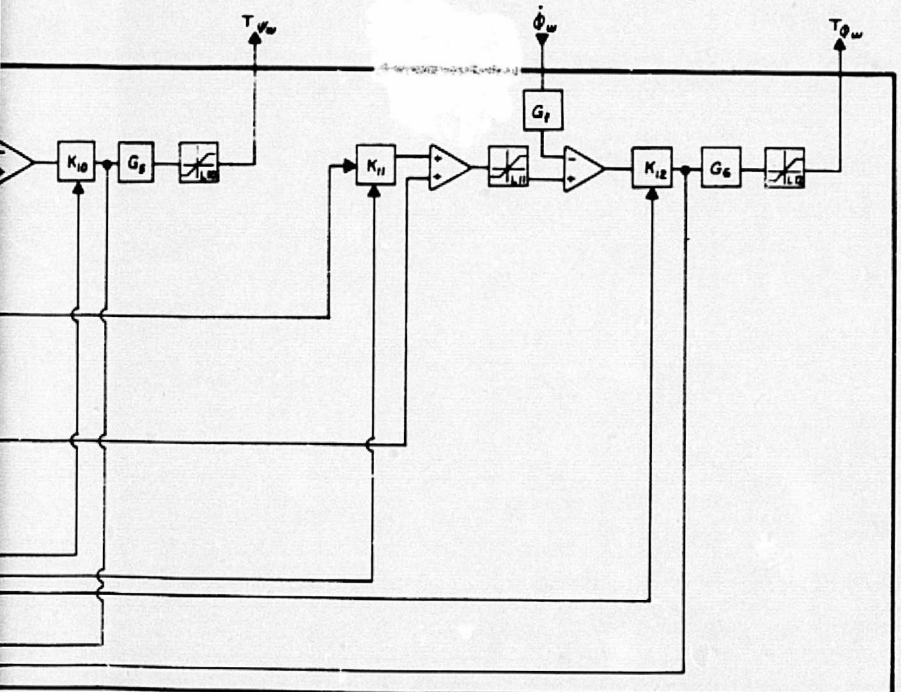


TROL LAWS AND SERVO SYSTEM (TRANSLATION)



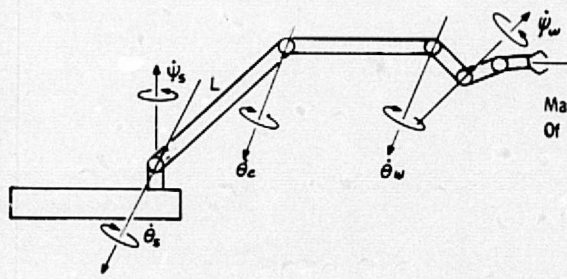
TROL LAWS AND SERVO SYSTEM (ROTATION)

Notation

1. $\dot\psi_s$, $\dot\theta_s$, $\dot\theta_e$, $\dot\theta_w$, $\dot\psi_w$, $\dot\phi_w$ = actual gimbal rate
2. $T\psi_s$, $T\theta_s$, $T\theta_e$, $T\theta_w$, $T\psi_w$, $T\phi_w$ = commanded gimbal torques
3. $\dot R$, $\dot A$, $\dot E$ = Range, Azimuth, & Elevation Commands
4. $\underline{\dot X_c}$ = X, Y, and Z commands given in terminal device axis
5. $$T = \begin{bmatrix} C(\theta_w+\gamma)C\psi_w & -C(\theta_w+\gamma)S\psi_w & S(\theta_w+\gamma) \\ S\psi_w & C\psi_w & 0 \\ -S(\theta_w+\gamma)C\psi_w & S(\theta_w+\gamma)S\psi_w & C(\theta_w+\gamma) \end{bmatrix}$$

   Terminal device to range vector transformation
6. $\gamma = 1/2\ \theta_e$
7. L = Lower arm segment length
8. $\dot\theta_{wc}$, $\dot\psi_{wc}$, $\dot\phi_{wc}$ = commanded wrist attitude rates
9. $$\dot\theta_{wh} = -\dot\theta_e - \dot\theta_s - T\psi_w S\theta_1\dot\psi_s$$
   $$\dot\psi_{wh} = -C\theta_1\dot\psi_s$$
   $$\dot\phi_{wh} = \frac{S\theta_1\dot\psi_s}{C\psi_w}, \text{ where } \theta_1 = \theta_e + \theta_s + \theta_w$$

   wrist attitude Hawk commands
10. K_i, i = odd = variable controller sensitivity gain
11. K_i, i = even = variable gimbal forward loop gain
12. L_i, i = odd = computed gimbal rate limit
13. L_i, i = even = computed gimbal torque limit
14. G_i, servo compensation network
15. G_f, tachometer ripple filter



Manipulator Degrees Of Freedom

FOLDOUT FRAME 2

(Eq. 9, Figure IIF-11) will provide fixed attitude control of the terminal device. Hawk mode commands will be used so that the operator can translate the manipulator in range, azimuth and elevation without incurring terminal device rotations. To implement Hawk mode control, the needed wrist gimbal attitude changes are computed from knowledge of changes in the shoulder and elbow gimbal angles and rates. The computed values (Eq. 9, Figure IIF-11) are then summed with the operator commands and applied to the wrist actuators to maintain attitude hold.

In Figure IIF-11, the limiters L_i, i = odd, control the magnitude of the derived gimbal rate commands and thus prevent the joint rates from exceeding designed values as the manipulator is extended to the extremes of its operating volume. To prevent permanent magnet demagnetization and commutation arcing resulting from excessive motor currents, limiters L_i, i = even, are provided to control the torque commands derived from large error signals. These limiters, in conjunction with current limiting on the drive power amplifiers, fully protect the dc torquers from exceeding any design parameter.

The gains K_1, K_3 and K_5 determine the translational controller sensitivity and are operator-variable. Likewise, the gains K_7, K_9 and K_{11} set the rotational controller sensitivity and are operator-variable. Gains K_2, K_4 and K_6 vary the translational motion servo stiffness and are adjustable from maximum values to zero. The zero setting allows the shoulder yaw, pitch, and elbow pitch gimbals to freely backdrive. Rotational servo stiffness is similarly variable from maximum to zero, thus permitting the wrist attitudes to easily backdrive and self-align. Filters G_f and G_i, i = 1, ... 6, are the tachometer ripple filters and servo compensating networks, respectively.

To summarize, the prominent features of the control system are:

1. Simple equations, no matrix inversions needed.

2. Manipulator applied forces and moments visually displayed to the operator.

3. Variable servo stiffness permitting "free" gimbal motion.

4. Range, azimuth, elevation, and X, Y, Z motion controllable in the spherical base and terminal device cartesian axis systems, respectively.

5. Easily servo compensated to accommodate large gain and inertia changes.

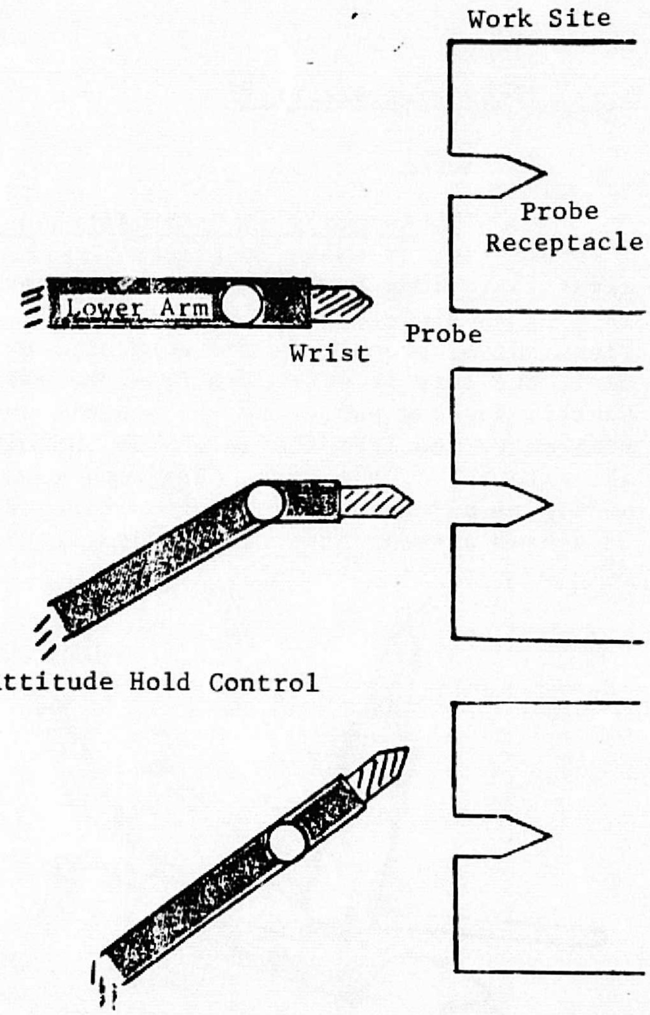
d. Axis Alignment - The operator has the option of selecting up to three different control axis alignments to facilitate performance of the various subtasks. For example, when the operator is doing gross translations such as moving a beam from the pallet to a point near final alignment he will be observing an overall view of the arm on his video monitor. Using a camera mounted near the manipulator shoulder, the operator will be most comfortable with a coordinate system in which range is in and out, azimuths left and right, and elevation up and down as viewed in the monitor. In the simulation task where a beam is removed from a pallet and the two ends are aligned in final position, two additional TV cameras are employed, one looking in each direction along the long axis of the beam. When the operator looks at one of these cameras he prefers his coordinate system reoriented to originate at the source of the scene--the left- or right-looking end effector cameras. In addition, because of the precise motions needed for final alignment, an "XYZ" system is preferable to the spherical range, azimuth, and elevation system.

This feature has been provided by the control laws and the operator has the option to select a control axis which corresponds to his primary monitor scene. This option was exercised by the three operators during the 90 data runs.

e. Attitude Hold - Attitude hold control refers to the automatic full or partial attitude hold of the manipulator wrist gimbals. With an attitude hold mode activated, the operator can translate the manipulator in range, azimuth or elevation and the wrist gimbals will be automatically driven such that the terminal device does not change its original attitude. Figure IIF-12 depicts an initial manipulator position with respect to a fixed work site followed by two final positions indicating how the wrist attitude (one DOF shown only) changed with and without attitude hold control.

To implement attitude hold control, the needed wrist gimbal attitude changes are computed from knowledge of changes in the shoulder and elbow gimbals. The computed values are then summed with the operator commands and applied to the wrist actuators to maintain the end-effector attitude in the desired position.

Range Attitude Hold - Two types of attitude hold control were used in the SMA simulation. The first technique, denoted



a. Initial Position

b. Final Position with Attitude Hold Control

c. Final Position without Attitude Hold Control

Figure IIF-12 Wrist Attached Hold Mode Example

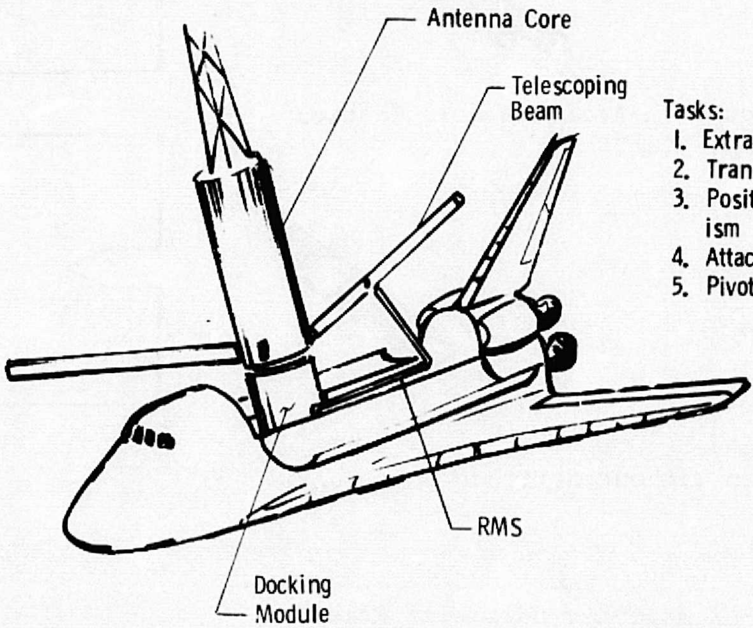
"range attitude hold", was the simplest of the two methods in that only the wrist pitch was affected. When a range translational command was given, a drive to the wrist pitch was applied to prevent an attitude change.

Full Attitude Hold - The second method was a "full attitude hold" in that all three wrist gimbals were driven to prevent an attitude change from a range, azimuth, or elevation translational motion. The three hawk commands were determined by computing the end-effector body rates, given the shoulder and elbow gimbal rates, and then deriving wrist gimbal rates from these body rate values.

5. Simulation Description

a. Tasks

1) Assembly of Radio Astronomy Telescope - The actual in-orbit task is shown in Figure IIF-13. It consists of attaching eight 55-ft long beams to a center telescoping core. This core is 8 ft in diameter and 45 ft long. It contains the telescope electronics, propulsion and antenna feed. Prior to beam attachment, the core is extracted from the cargo bay and placed on the Shuttle docking port with the Shuttle manipulator. Each beam is then extracted from the cargo bay and attached, in sequence, to the exterior of the core. The core must be rotated on the docking module to allow the manipulator to reach each beam attach point. This beam attachment task was simulated.



Tasks:
1. Extract Beam From Cargo Bay
2. Translate & Align
3. Position at Attach Mechanism
4. Attach Upper Mechanism
5. Pivot to Lower Attachment

Figure IIF-13 Radio Astronomy Telescope Beam Placement with RMS

The Shuttle manipulator is presently 50 ft long. Our SMA has a 13-ft reach. The mockups are 1/4 scale to be compatible with the 13/50 ratio between the SMA and the Shuttle payload handling systems. The mockups shown in Figure IIF-14 consist of a portion of the Shuttle cargo bay, a center telescope core with a female beam attachment mechanism, and a beam with a male attachment mechanism. (The drawings for these mechanisms are shown in Appendix C.) These mockups were positioned around the SMA to

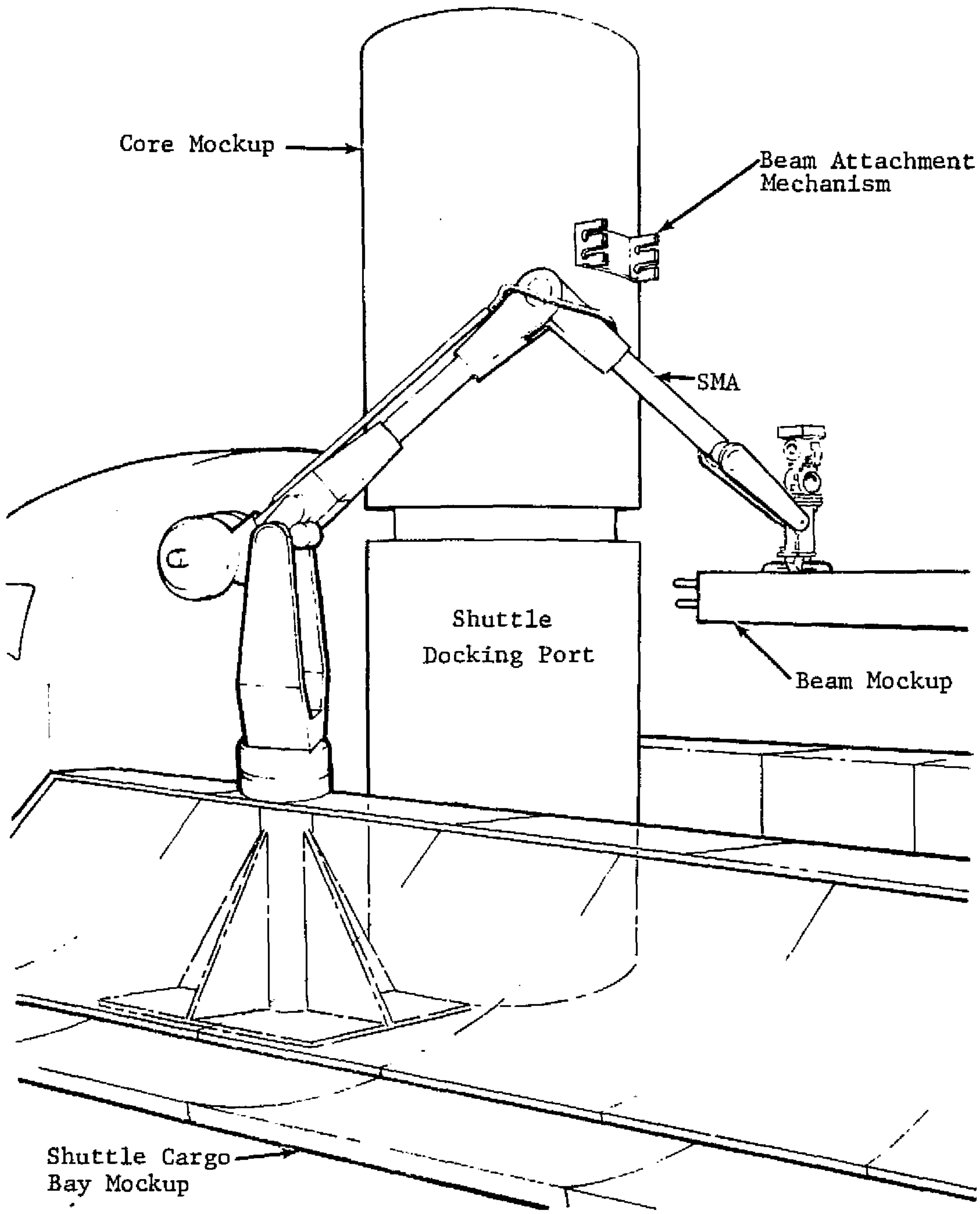


Figure IIF-14 Radio Astronomy Telescope Simulation

represent the actual Shuttle dimensions as closely as possible.

Video camera placement was one of the variable test parameters. Their locations were constrained to points within the cargo bay where cameras can actually be located.

Two video cameras were utilized for this simulation phase. There was no direct vision of the task by the operator. One camera, located on the starboard cargo bay door hinge line, was mounted on a pan/tilt unit. This camera viewed the overall cargo bay to monitor the major beam translations. We utilized a 10 mm wide-angle lens which provided a complete view of the beam as it was translated. The view from this camera was also utilized during the beam attachment phase. This provided a side view which gave the operator closure distance and rate data. The ability to increase the focal length of this camera is desirable and later simulations should use a remotely-controlled zoom lens. The actual zoom range will be determined by the physical dimensions of the final RAT design and its location in the Shuttle. However, based on our scaled mockups, a 10 to 50 mm zoom lens would be desirable. We recommended that this camera be located on a pan/tilt unit, mounted near the starboard cargo bay hinge line, perpindicular to the beam attach point on the RAT core.

The second camera was located under the RAT beam and would (operationally) be mounted on an end effector jaw. This position allows a view of the lower two attachment pins, the female attachment device and the standoff-cross alignment device. A 25 mm lens was used on this camera. A light was initially attached to this camera to illuminate the alignment standoff cross. This light was removed, since we found external flood lights were adequate and created less disturbing shadowing.

Initially, the beam attachment task was evaluated without a standoff cross to aid alignment. We soon found that the beam positioning and alignment task was quite difficult due to the lack of visual areas, however, the operators were still in their learning curve. A scaled, Apollo type standoff cross was mounted below the attachment device. It was mounted so that when correctly aligned, the male beam attachment pins were in turn aligned and almost touching the female receptacle on the core. The attachment mechanisms and the standoff cross are shown in Figures IIF-18 and IIF-20. Translational accuracies required for this alignment were $\pm.125$, $\pm.0312$, and $\pm.125$ inch for X, Y, and Z, respectively. The attachment mechanism is made of steel. Initially the glare off the unpainted steel made identifying the receptacle

grooves difficult to see through the side mounted video systems. The female attachment receptacle was painted matt black and a portion of the front receptacle was painted white. The male attachment receptacle, on the beam, was also painted black and the alignment pins white. The paint scene, shown in Figure IIF-18, aided significantly in the final alignment task by providing clearly definable contact surfaces.

The operators were initially used to examine and develop various test setups. Various camera positions, manipulator control modes, lighting, alignment aids, etc. were tried. This allowed the operators to become completely familiar with the simulation, allowed suggestions to be incorporated and decreased the final learning curve time. The general task sequence is shown in Figures IIF-15 thru IIF-21. Translation and coarse alignment were not a control problem. The video monitor mounted on the cargo bay higne line generally provided adequate visual feedback to the operator for these tasks. The full automatic attitude hold control mode kept the beam attitude constant during translations. Final alignment requires attitude and position alignments to within $\pm$ 1/16". This fine alignment is probably more critical than would be designed into a space system. However, our total system provided operator control which allowed this task to be successfully completed.

The final alignment task consisted of first aligning the lower attachment pins with the receptacle and then translating forward until the pins contact the receptacle. The end effector camera is used to align yaw and roll attitude and Z position. The side camera is used to maneuver the beam in pitch and X position. Once the pins contact and alignment is verified, they are then positioned to the bottom of the receptacle groove. As the pins go into the receptacle groove, retainers are activated which hold the bottom pins in place. The second phase of the attachment is to pitch the beam up which rotates the upper attachment pins into place where they are also retained. This pitch maneuver could not be accomplished without setting the manipulator wrist torques to zero, which allowed them to backdrive as a +Z translation command was made. If the beam attachment mechanism design, which is proposed in this study, is pursued to a space application then its attachment characteristics will probably impose a design requirement on the manipulator that provides the operator with a selectable wrist torque output.

The antenna beam assembly task was broken into the following subtasks for the simulation:

1. Extract beam from cargo bay.
2. Translate beam to core attachment interface.
3. Position and align beam (male) attach mechanism at core (female) attach mechanism.
4. Maneuver upper (2) beam attach pins into core receptacles and verify placement.
5. Pitch beam to engage lower two pins and verify.

Data is discussed in Section 6.

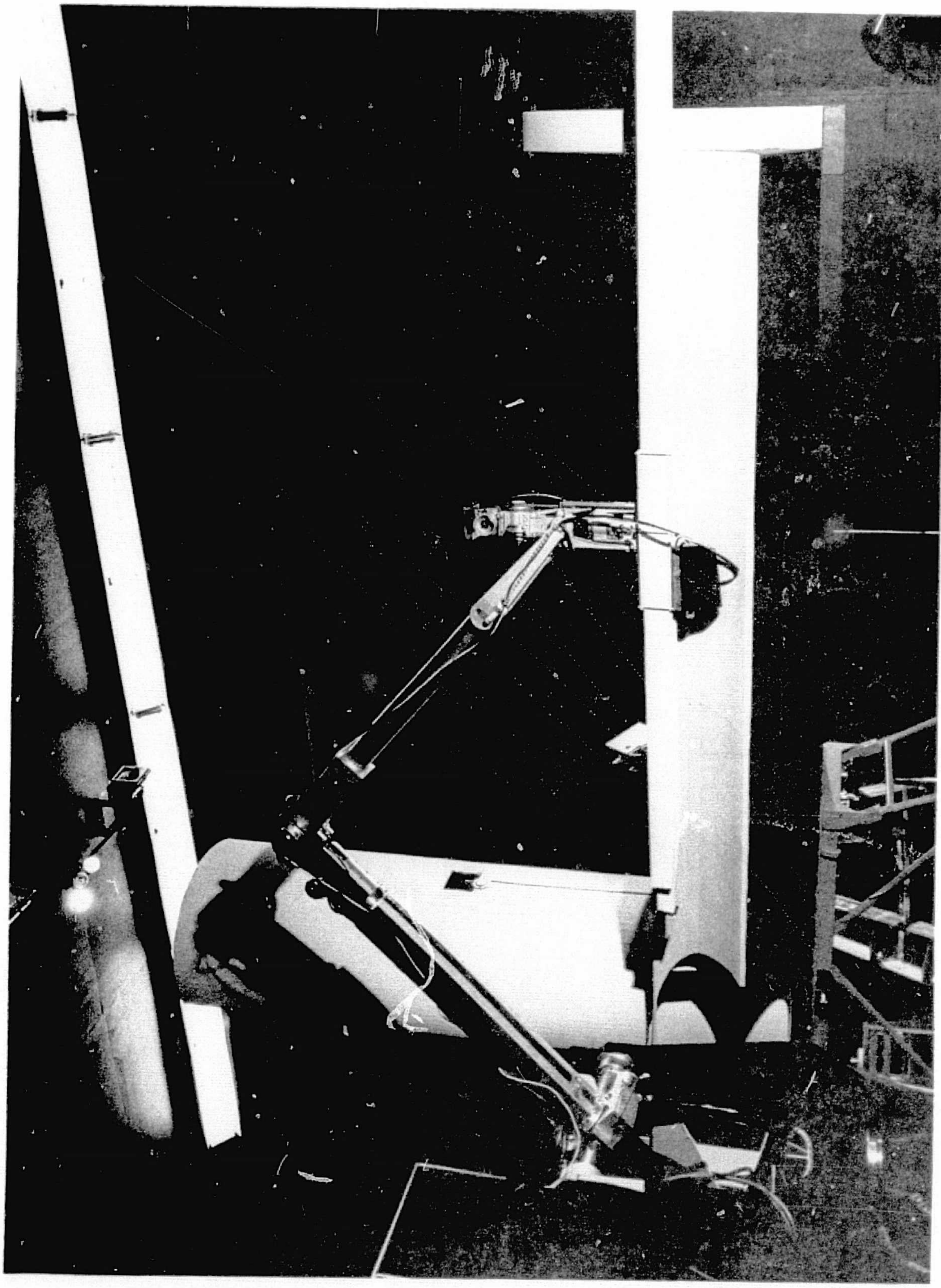


Figure IIF-15 RAT Assembly Simulation - Starting Point

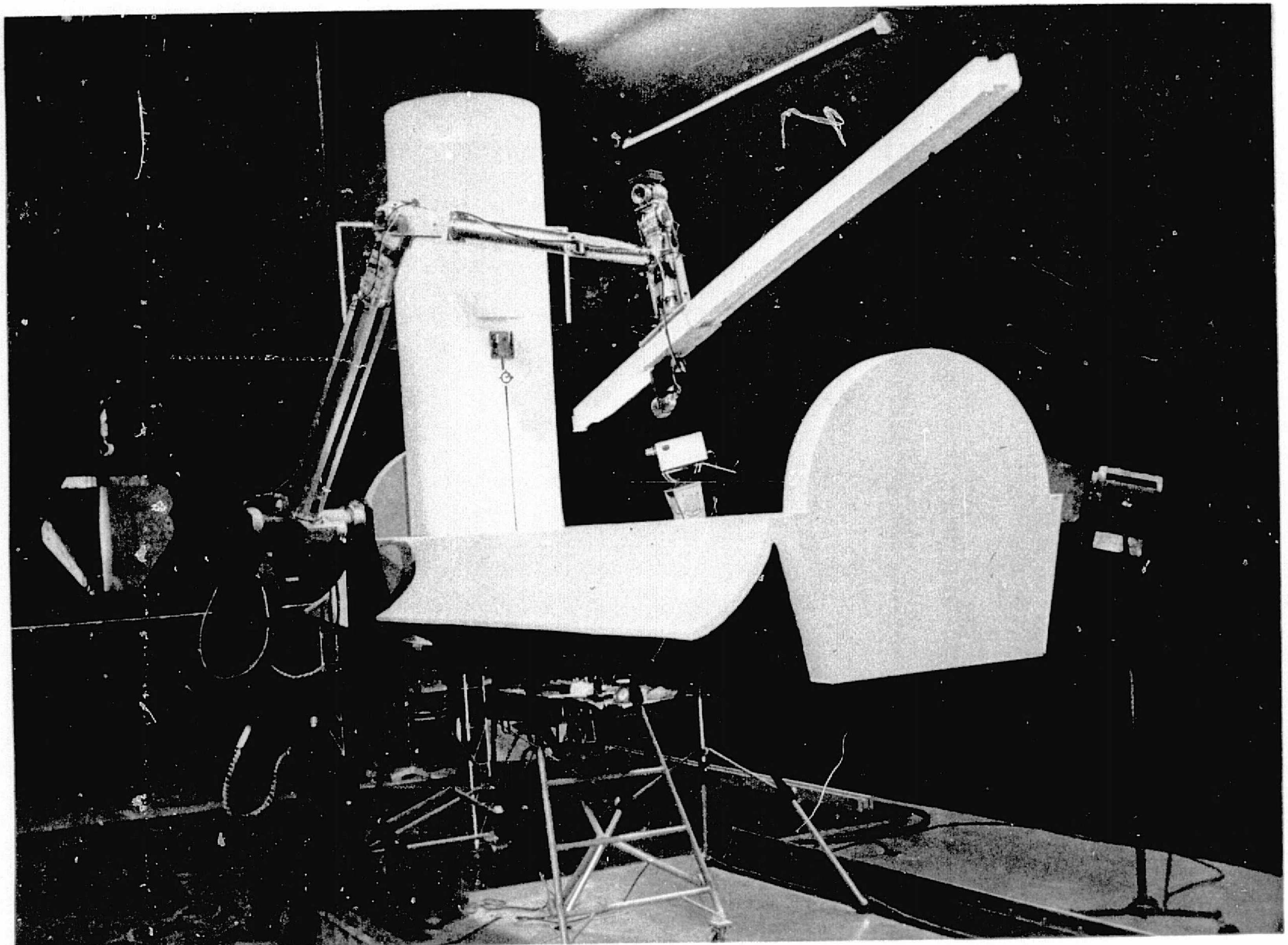


Figure IIF-16 RAT Assembly Simulation - Beam Translation

Figure IIF-17 RAT Assembly Simulation - Attachment Alignment

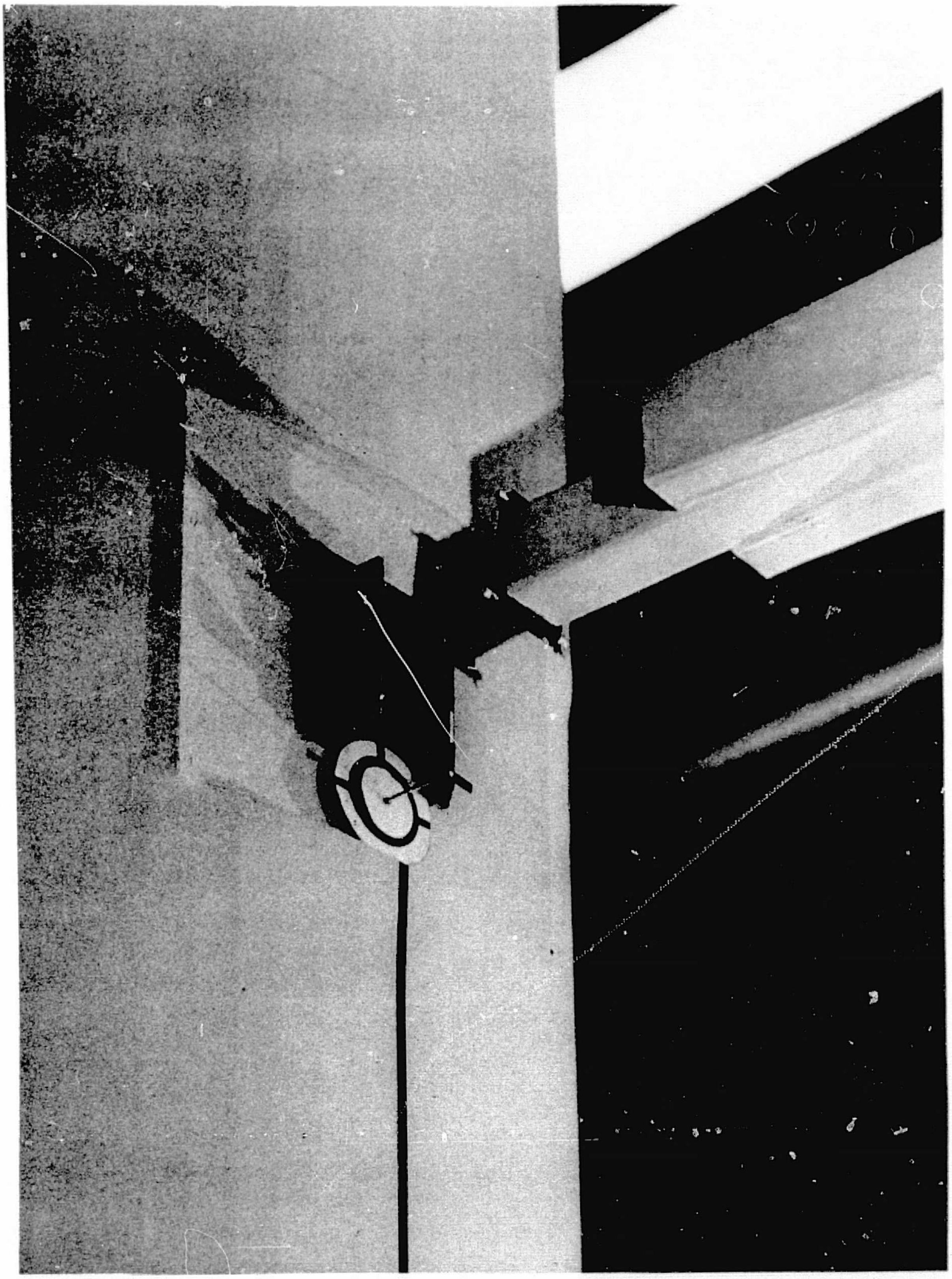


Figure IIF-18 RAT Assembly Simulation - Attachment/Alignment Mechanisms

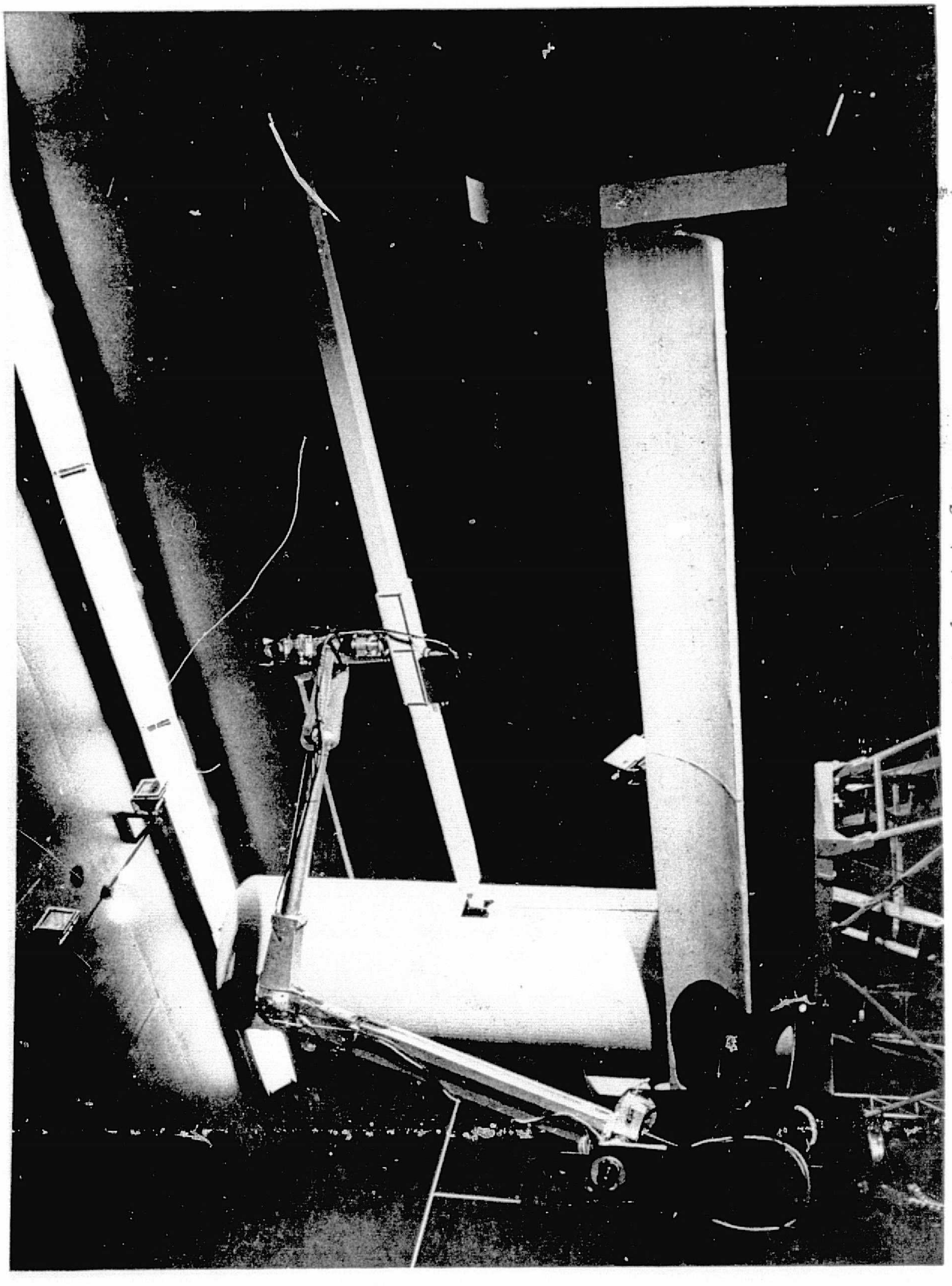


Figure IIF-19 RAT Assembly Simulation - Beam Attachment to Core

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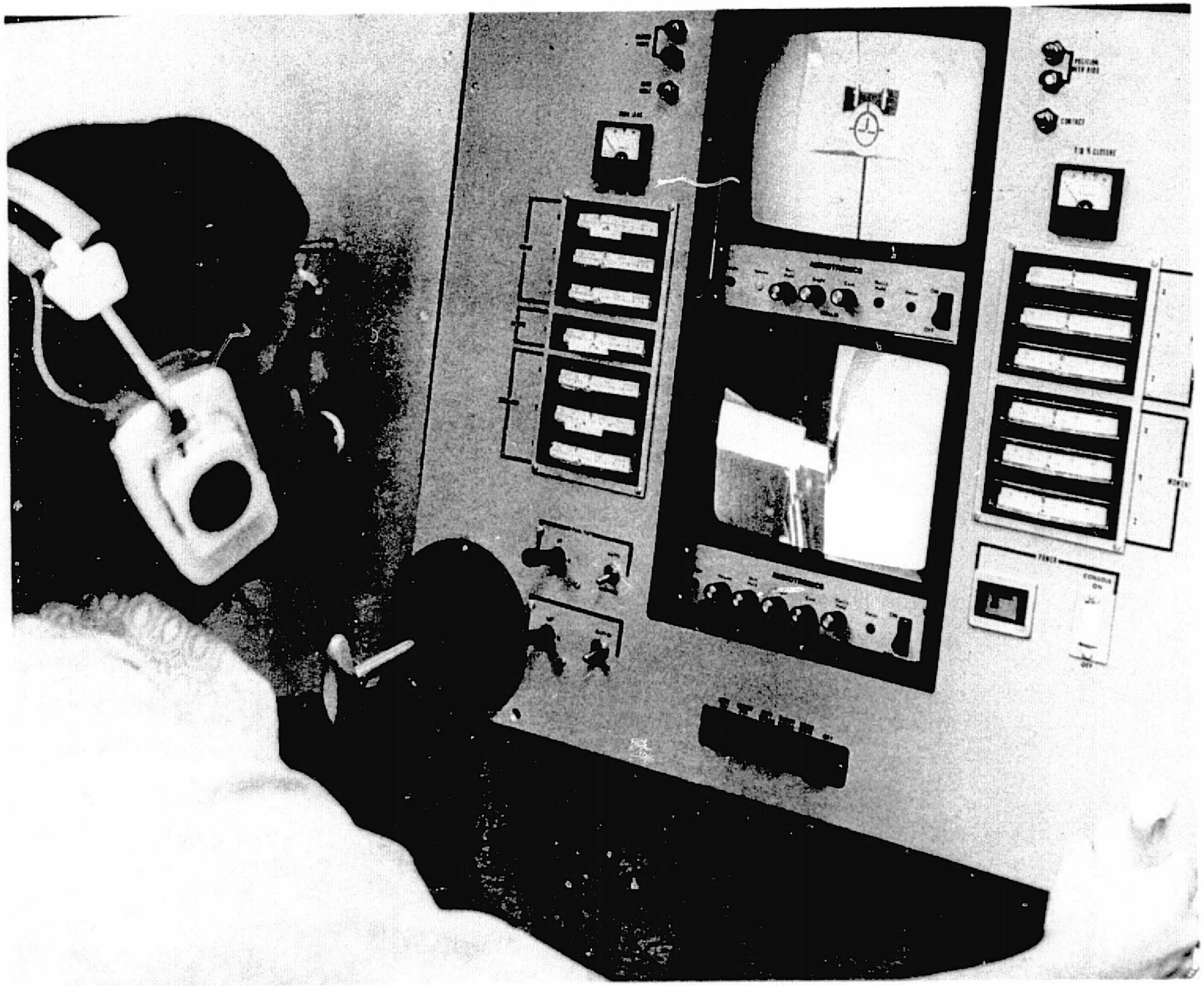

Figure IIF-20 RAT Assembly Simulation - Control Console/Beam Attached

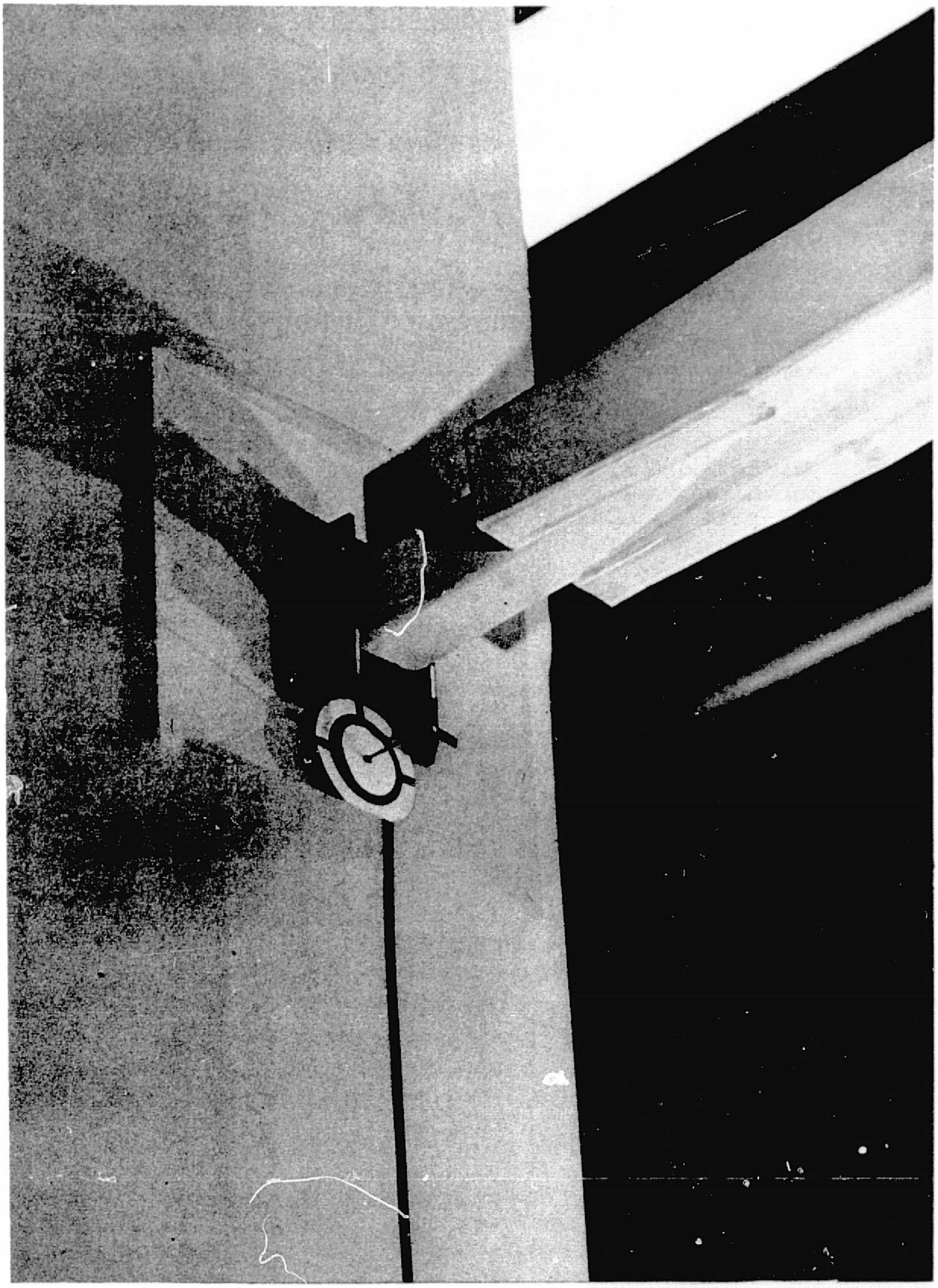

Figure IIF-21 RAT Assembly Simulation - Beam Attached

2) Assembly of Microwave Power Transmission System (MPTS)

The assembly of a total MPTS represents a massive in-space assembly task. The one-kilometer transmission antenna structure is composed of thousands of 60-ft long beams which are all attached at their ends. This building block design contains assembly tasks which are highly repetitious. There are two basic beams: a triangular (horizontal) type which is 30 in. x 60 ft., and a square (vertical) type which is 30 in. x 60 ft. Our assembly concept utilizes a 70-ft. long (reach) mobile assembler (MA) for structural assembly. This MA maneuvers along the structure, uses a 7 DOF manipulator to build the structure and is controlled remotely from the ground. The basic in-space task is to extract beams from the docked beam pallet, translate them (up to 120 ft. distance) to their proper locations, align and hold the end(s) until rigid attachment is made. This concept is shown in Figure IIF-22.

Our mockups, shown in Figure IIF-23 are approximately 1/5 scale, sized by the 13/70 ratio between the SMA and the assembler manipulator. The mockups are composed of 12 beam segments and four cross-braces. This represents the upper segments of two 60-ft. squares. The rear square is made up of solid members to simplify mockup construction. The forward square beams are constructed of tubular members which are properly scaled and each is removable. Video cameras are placed on the corners of the beam structure, to the right and left of the manipulator shoulder. This corresponds to the actual design which locates these cameras on the mobile assembler base. Each camera is mounted on a remotely controlled servo-driver pan/tilt unit which will either auto-track the manipulator wrist or be driven in a manual mode. These cameras had 10mm lens. A second set of video cameras were mounted on the manipulator end effector. These fixed cameras view each end of the beam and aid in final alignment in task 1. Only one of these cameras was used for task 2. The SMA was also axis-aligned for these cameras. The SMA motions, remotely controlled from the SMA control station, are axis-aligned with the video camera selected for control. This feature automatically aligns the manipulator translational motions with the video image seen by the operator. Standard Apollo-type rate controllers were used to control the SMA.

This simulation series consisted of placing the two horizontal (triangular) beam segments which make up the upper and lower end caps of the antenna support structure. Task 1, Figure IIF-22(1), was to extract the beam segment from the beam pallet, translate and rotate the beam to the installation site, position and align both ends and contact the alignment plates while maintaining alignment. The second task, Figure IIF-22(2), was to install the triangular beam which is located 90 deg to the task 1 beam. This task differs significantly from task 1 in that the beam must be butted end to end rather than an overlay type task.

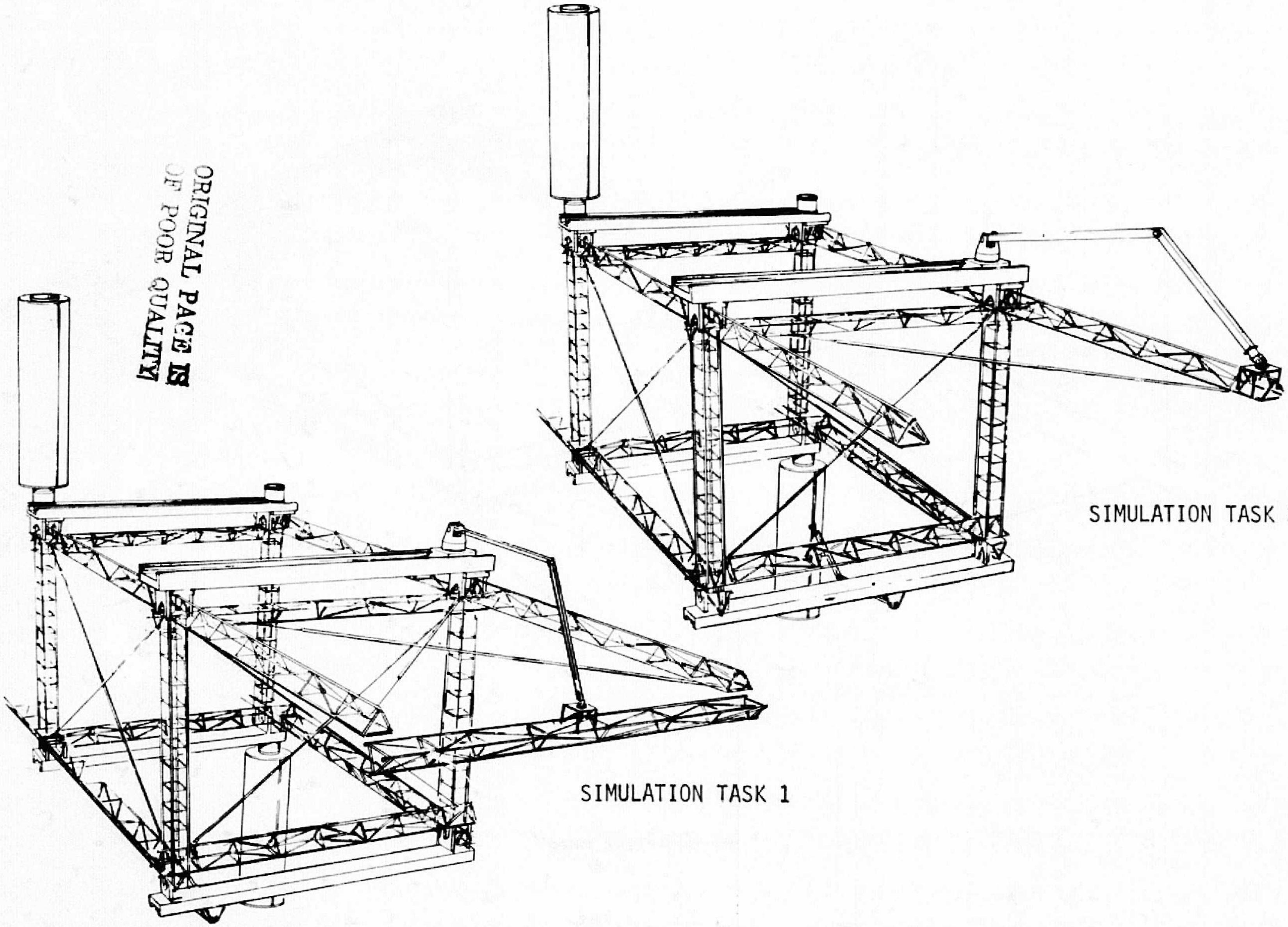

Figure IIF-22 Cube Assembly Sequence

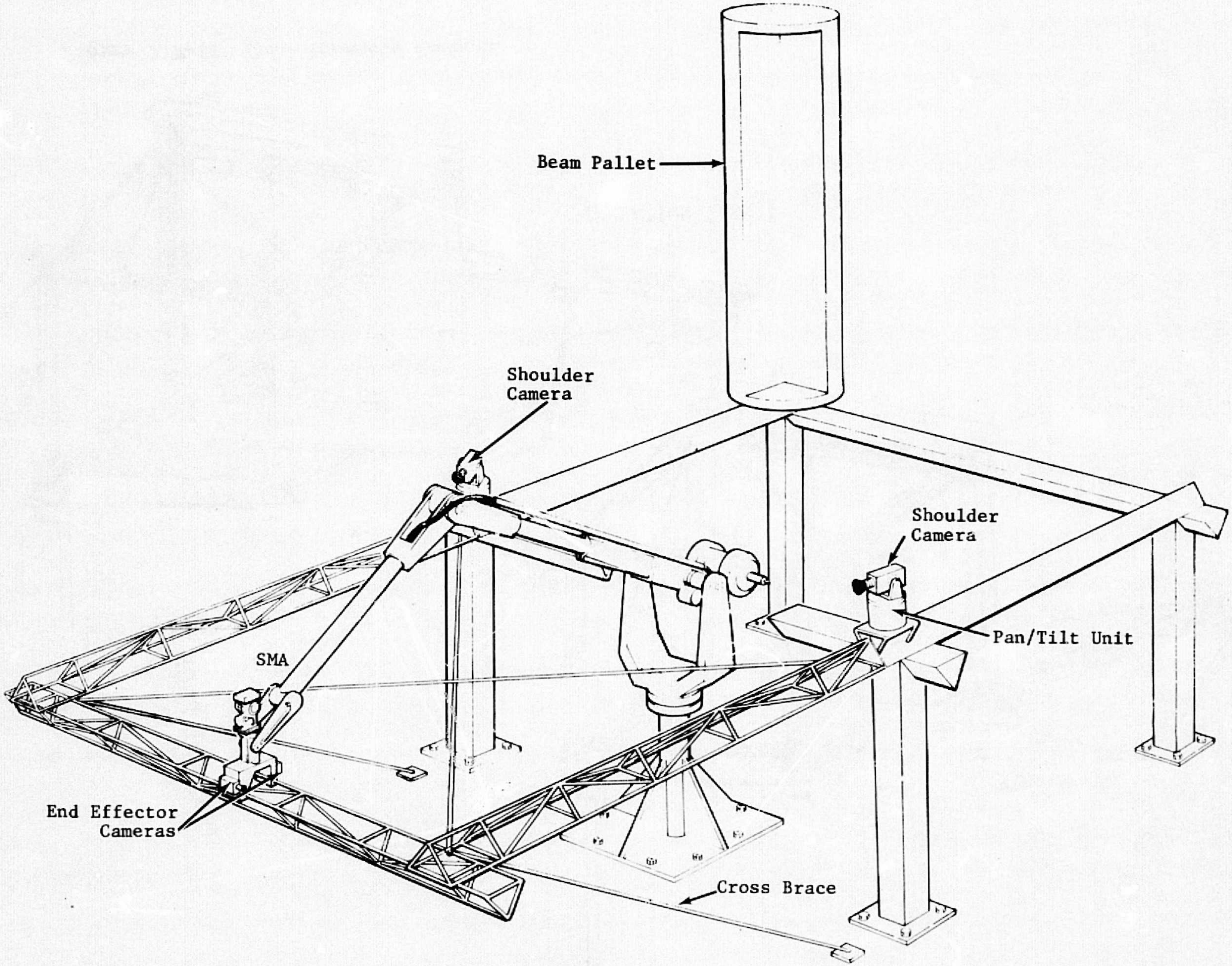

Figure IIF-23 Microwave Power Transmission System Mockup

MPTS Assembly - Task 1

The actual simulation sequence for task 1 is shown in Figures IIF-24 through IIF-30. The task started with the beam placed in the pallet mockup. The operation used the left shoulder camera to view the task and align the manipulator control axis. This included the beam extraction from the pallet, the major translation and the coarse alignment of the beam ends. The right shoulder camera was used to monitor the beam location and followed the translation in an auto-track mode. The operators generally translated the beam at approximately 0.5 ft/sec actual (2.5 ft/sec scaled up). The "range" wrist attitude hold mode was used during this phase. All of the operators learned this task quickly and were able to complete this phase in approximately 2 minutes (average) or approximately 50% of the total task time. During the major translation, the beam was rotated 90° in preparation for the alignment phase. This rotation is shown in Figures IIF-26 and IIF-27.

As the beam was positioned over the ends of the existing beams, Figure IIF-28, some indication of the manipulator extension in the "X" direction is required. Our control console contains joint angle meters which display the angle ($\pm$) of each of the manipulators joints. The shoulder and elbow pitch meters were marked when the manipulator was extended adequately to allow the beam to be brought down into place at the end of the existing beams. These reference marks were then used to verify adequate extension. The shoulder cameras, Figure IIF-28, viewing the opposite corners of the structure were used for the course beam alignment.

During the fine alignment phase, Figure IIF-29 and IIF-30, the operator switched to the end effector cameras and used the right facing camera for axis alignment. This subtask time also averaged approximately 2 minutes and took approximately 50% of the total task time. Our initial set-up placed the end effector cameras to the right and left of the beam. This configuration was quickly abandoned since the views were of opposite sides of the beam and presented some confusion to the operator when beam yaw motions were made. The cameras were placed in an end-to-end configuration, Figure IIF-29 and viewed mirrors which were located in the ends of the existing beams. These mirrors are located on the centerline of the cameras visual field and positioned at a 45° angle which provides the operator with a scene looking straight down at the tips of the beam being emplaced. A set of cross-hairs were mounted under the mirrors. The beam ends contained the target cross with which the cross-hairs were aligned. The beam was considered aligned when both sets of cross-hairs were centered in the target cross. Figure IIF-30 shows the operator's video monitors at the point of final alignment. This alignment required positional accuracies less than 1/8 in.

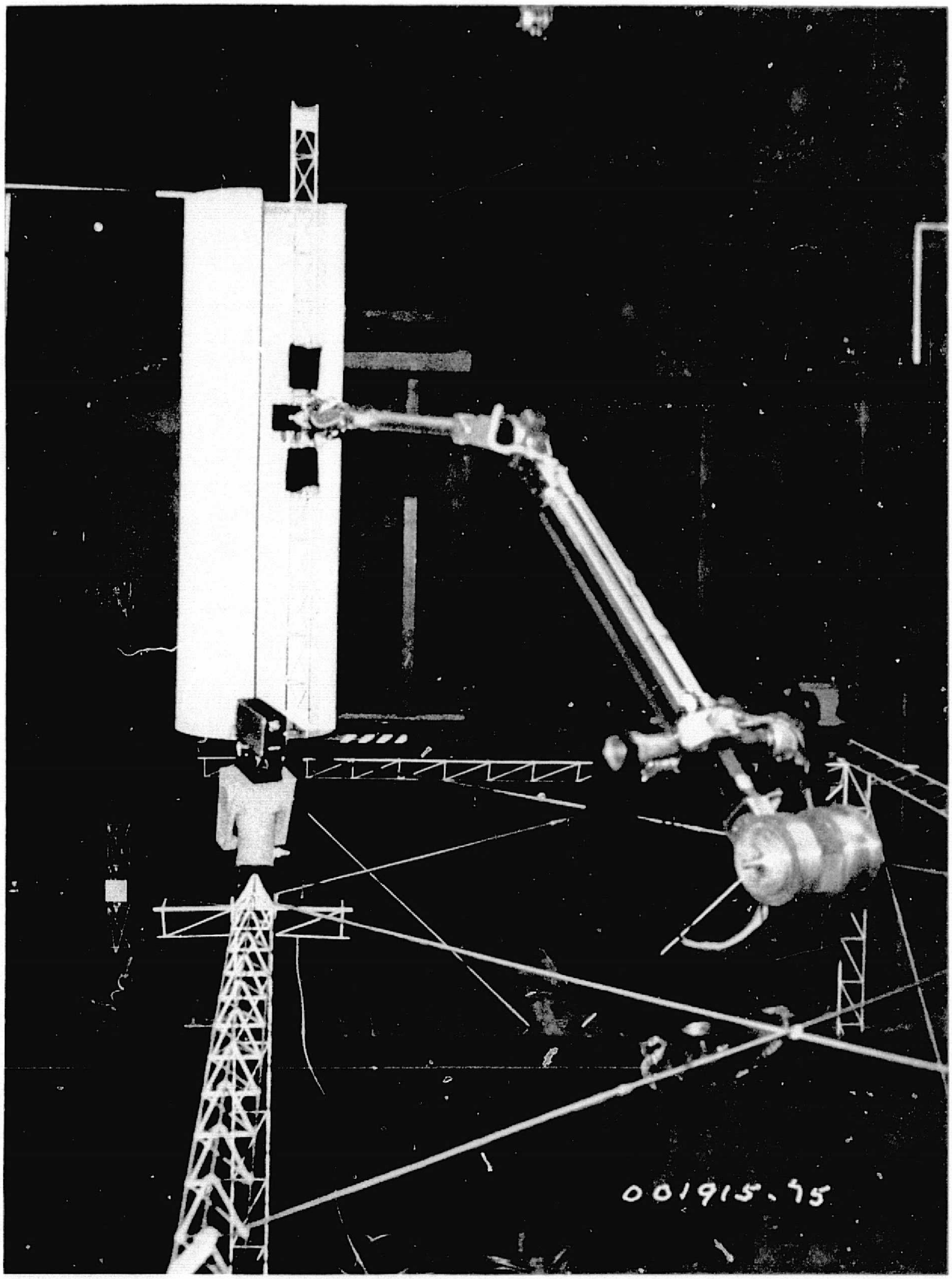

Figure IIF-24 MPTS Assembly Simulation, Task 1 - Beam Extraction, Wide View

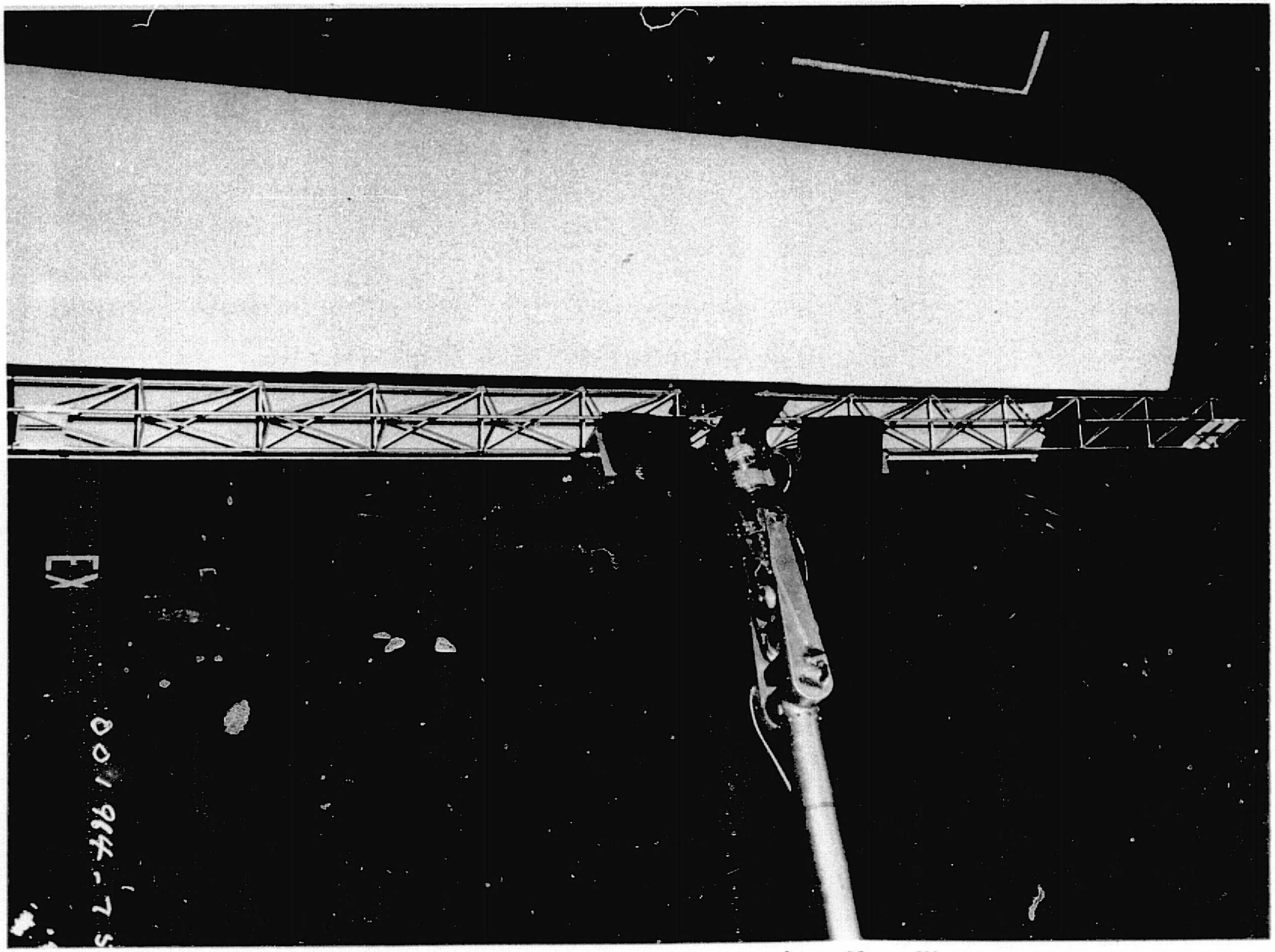

Figure IIF-25 MPTS Assembly Simulation, Task 1 - Beam Extraction, Close View

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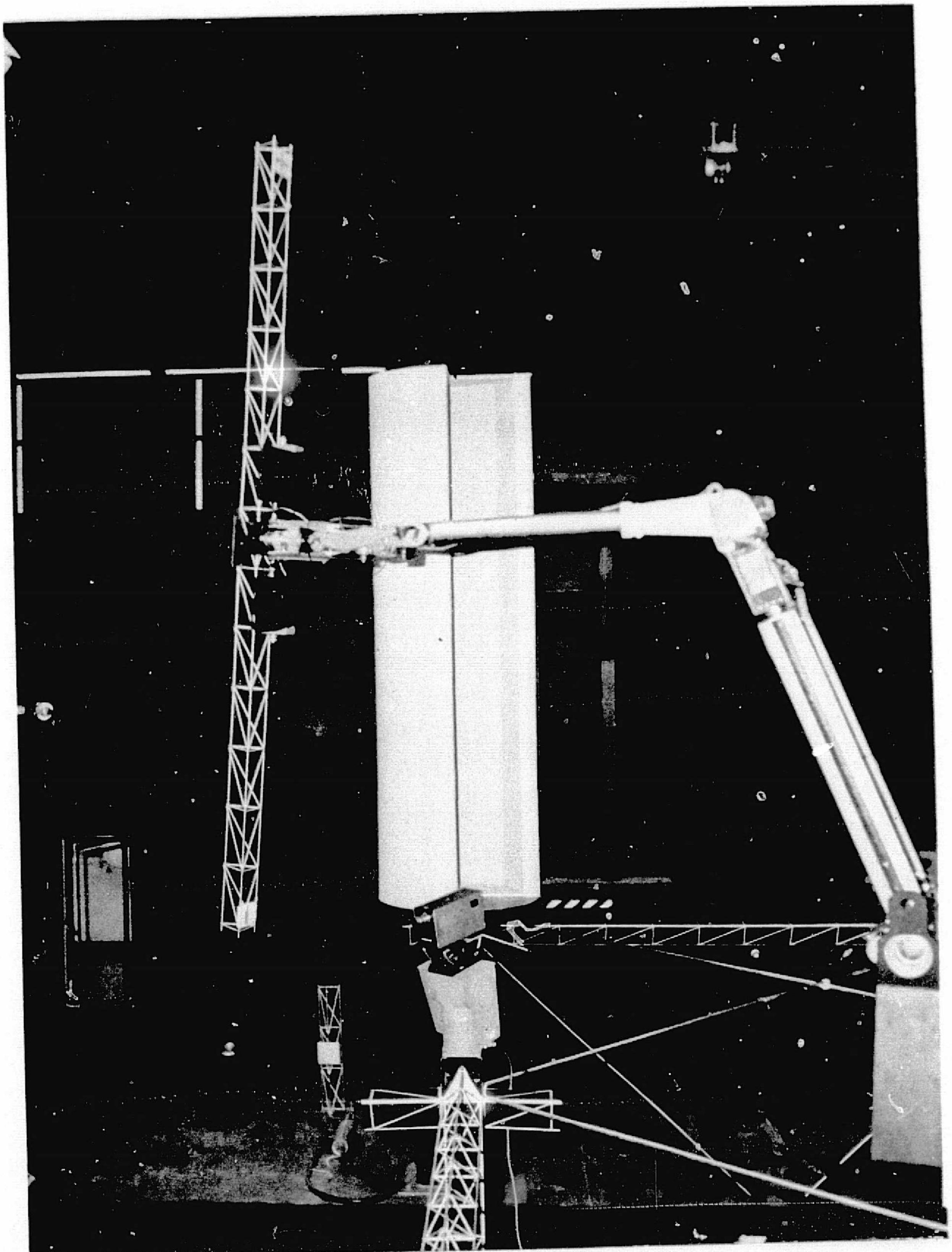

Figure IIF-26 MPTS Assembly Simulation, Task 1 - Beam Translation

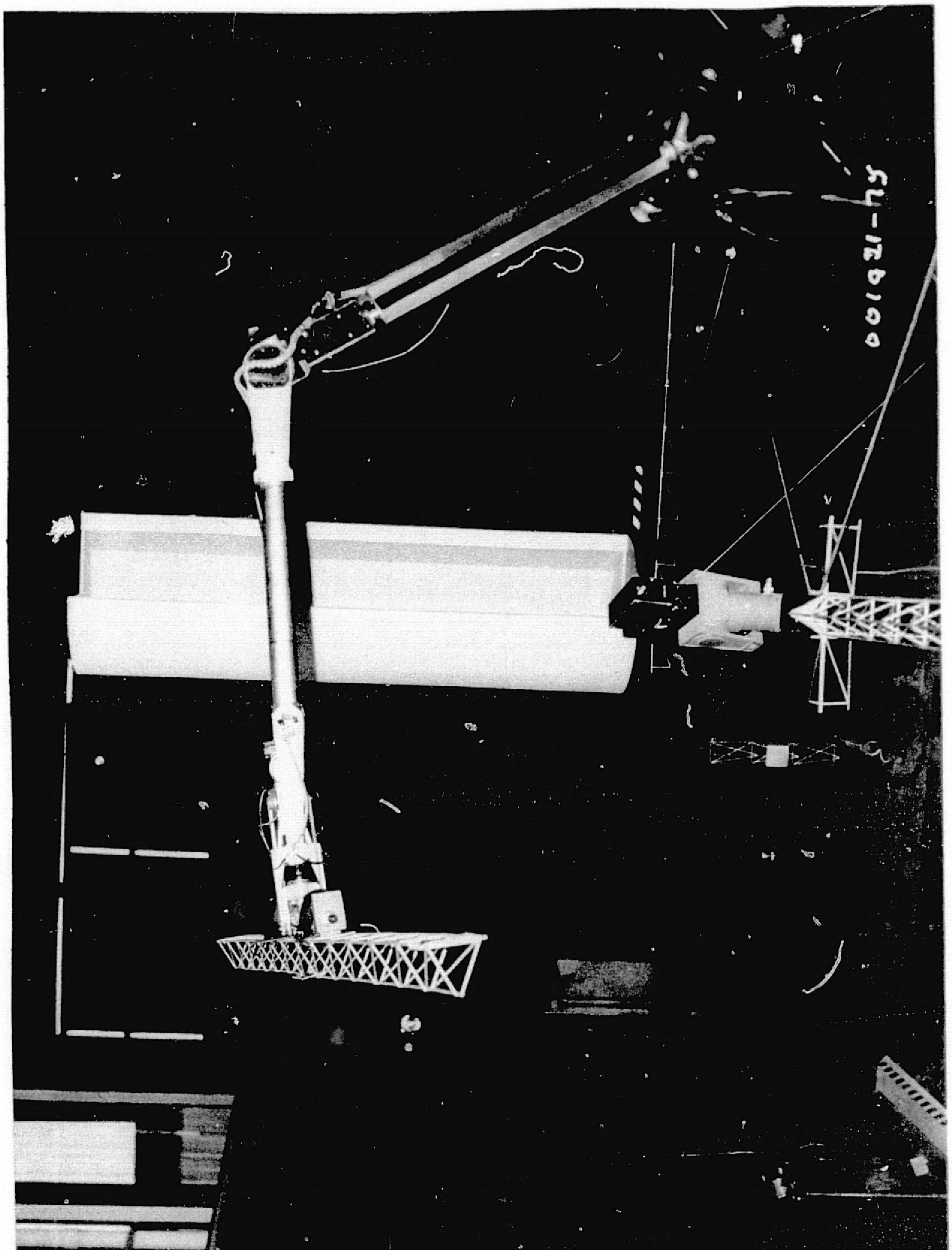

Figure IIF-27 MPTS Assembly Simulation, Task 1 - Beam Translation and Rotation

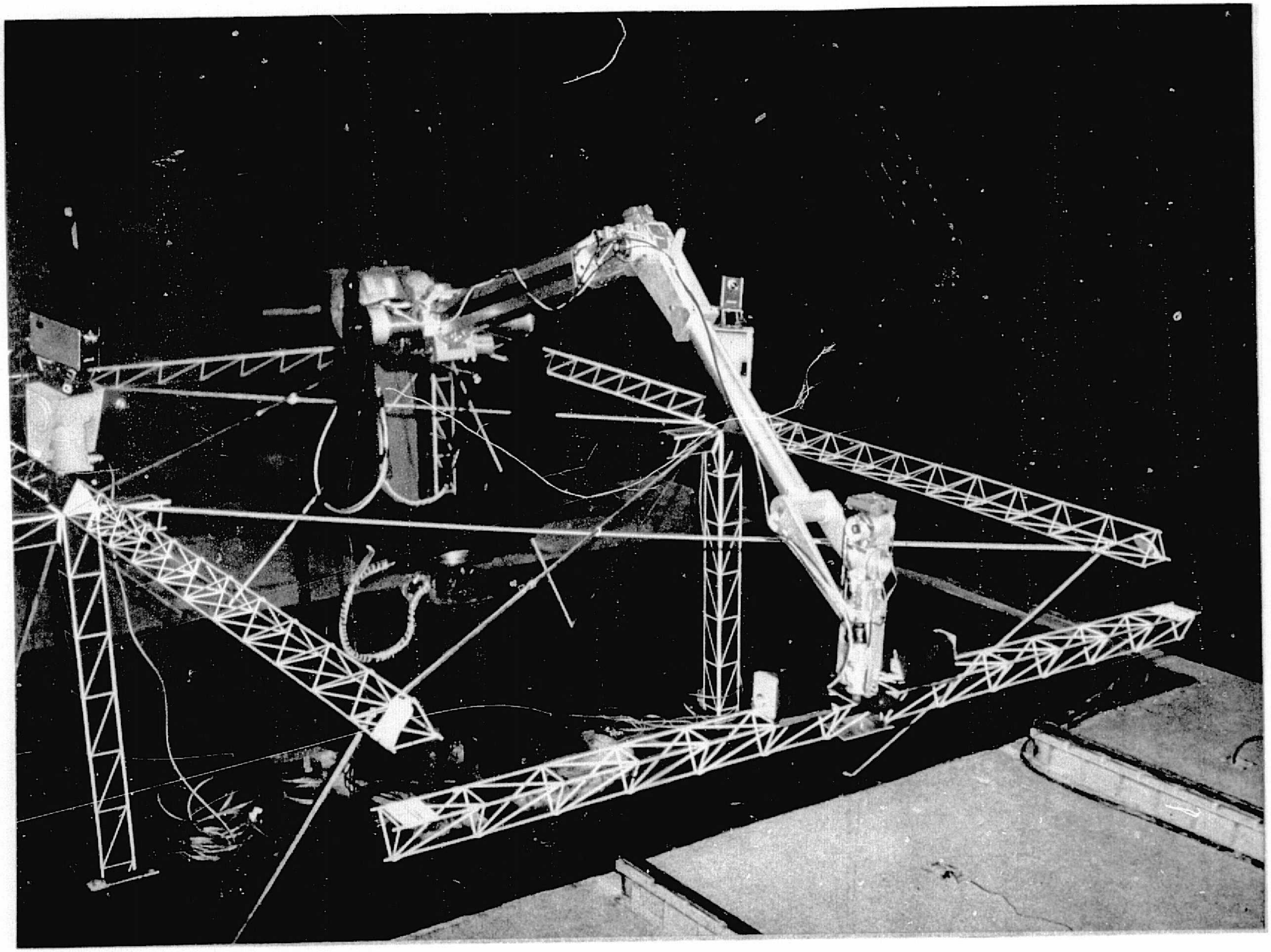

Figure IIF-28 MPTS Assembly Simulation, Task 1 - Beam Initial Alignment

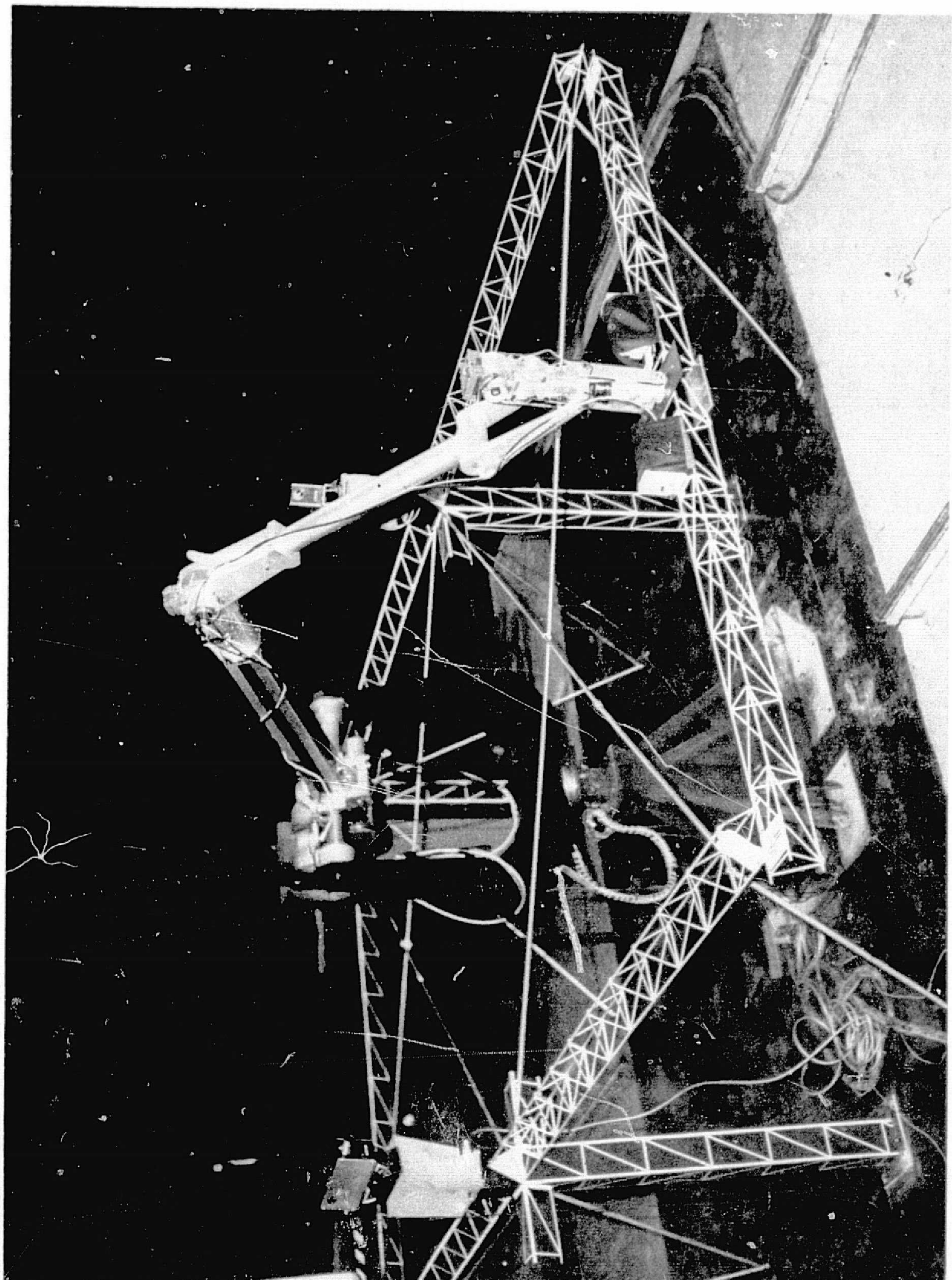

Figure IIF-29 MPTS Assembly Simulation, Task 1 - Beam Fine Alignment

Figure IIF-30 MPTS Assembly Simulation, Task 1 - Beam Fine Alignment, Video View

A second approach was evaluated for final beam alignment. This approach was evaluated subjectively, no actual operator performance data was taken. This concept utilized the mirrors and cross-hairs, however, a pin was installed on one beam end which was inserted into a receptical on the existing beam end. This pin was located on the right end and when inserted, provided a point at which the movable beam could be pivoted into place on the opposite end by turning the wrist torques to zero and simply translating the opposite end into place. This technique eliminates the need to align two unattached beam ends at the same time. Further evaluations are needed, however, at this point, an alignment concept similar to this would be recommended for an operational technique.

No significant problems arose during the double ended beam placement task. Detailed test results are discussed in Section 6.b.

MPTS Assembly - Task 2

The second assembly task in the MPTS series is to translate, align and attach a triangular beam which is assembled end to end rather than an overlay type as was used in task 1. This task started at a position midway between the pallet and the attachment point. Extraction from the pallet was demonstrated in task 1 and, therefore, would have been redundant for task 2. This beam segment was shortened to 4 ft. rather than use the full length since the manipulator end effector grasp the beam 2 ft. from the end to be attached. If a full beam was used, a large offset would be created in the SMA wrist yaw gimbal. This would be difficult to balance and would serve no purpose, since the video camera views the short end which contains the attachment pins.

The manipulator shoulder video cameras were located in the same position and had the same characteristics as were used in task 1. The third camera was located under the beam, on the SMA wrist centerline as shown in Figure IIF-31. This camera would be located, operationally, on an end effector jaw. The camera used in the simulation contained a 10mm wide angle lens.

The major beam translation and course alignment was made by viewing through the left shoulder camera, which tracked the end effector. The operators chose to axis align with this camera and maneuver in the "range" attitude hold mode. The translation and course alignment phase were readily accomplished in less than one minute. As the beam was rotated around into position, the operators selected the end effector camera for the prime view and axis alignment. A standoff cross was used initially for this task, but was abandoned when it was found that the beam ends could be used more effectively.

The right shoulder camera was also used for final alignment. Figure IIF-32 shows the operator's video monitor views during final alignment. The right shoulder camera views down on the beam intersection, Figure IIF-31, and provides the operator with X translation and yaw attitude data. Beam pitch is monitored intermittently through the left shoulder camera. The other beam axis are monitored through the end effector camera.

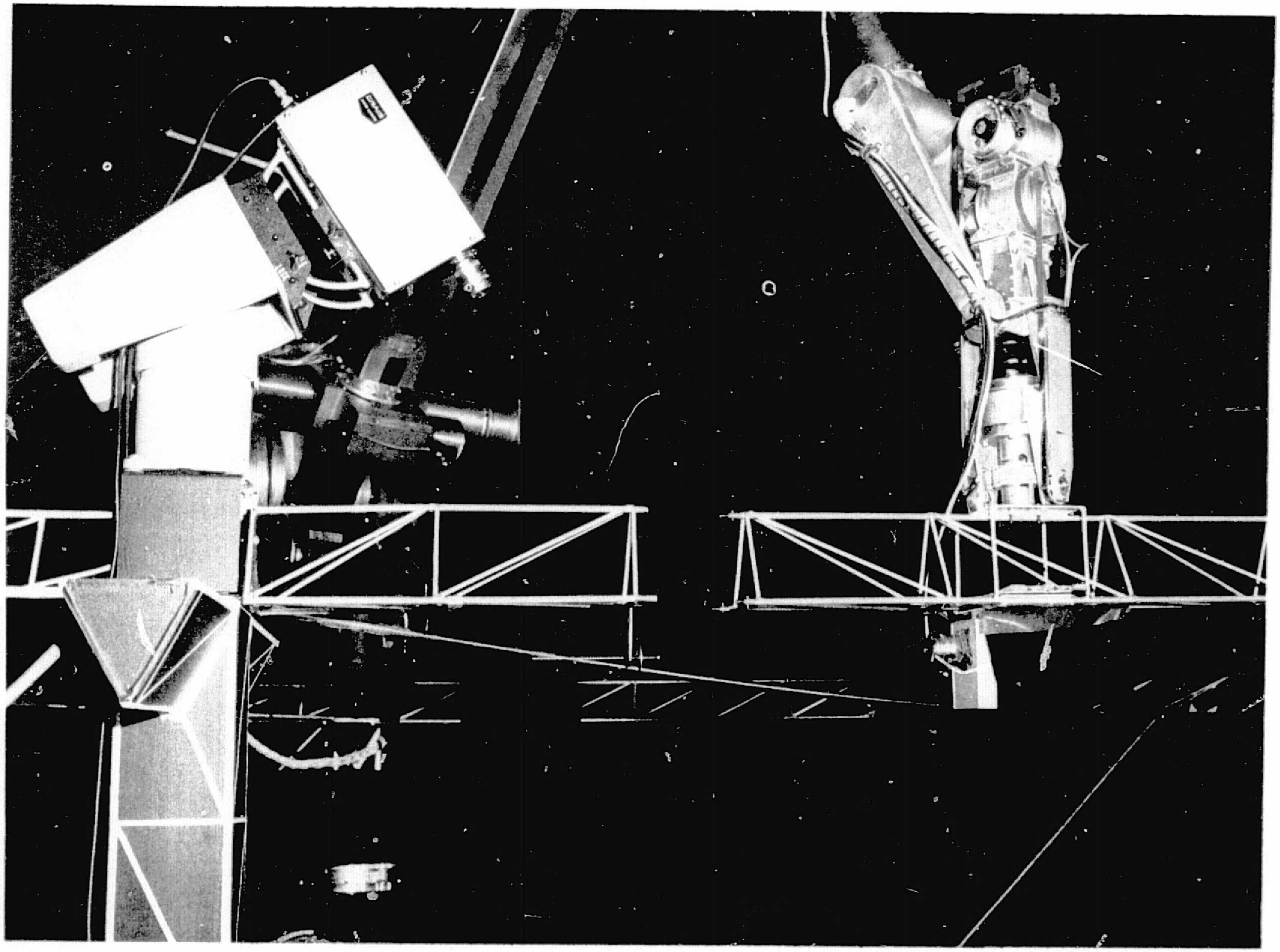

Figure IIF-31 MPTS Assembly Simulation, Task 2 - Beam Initial Alignment

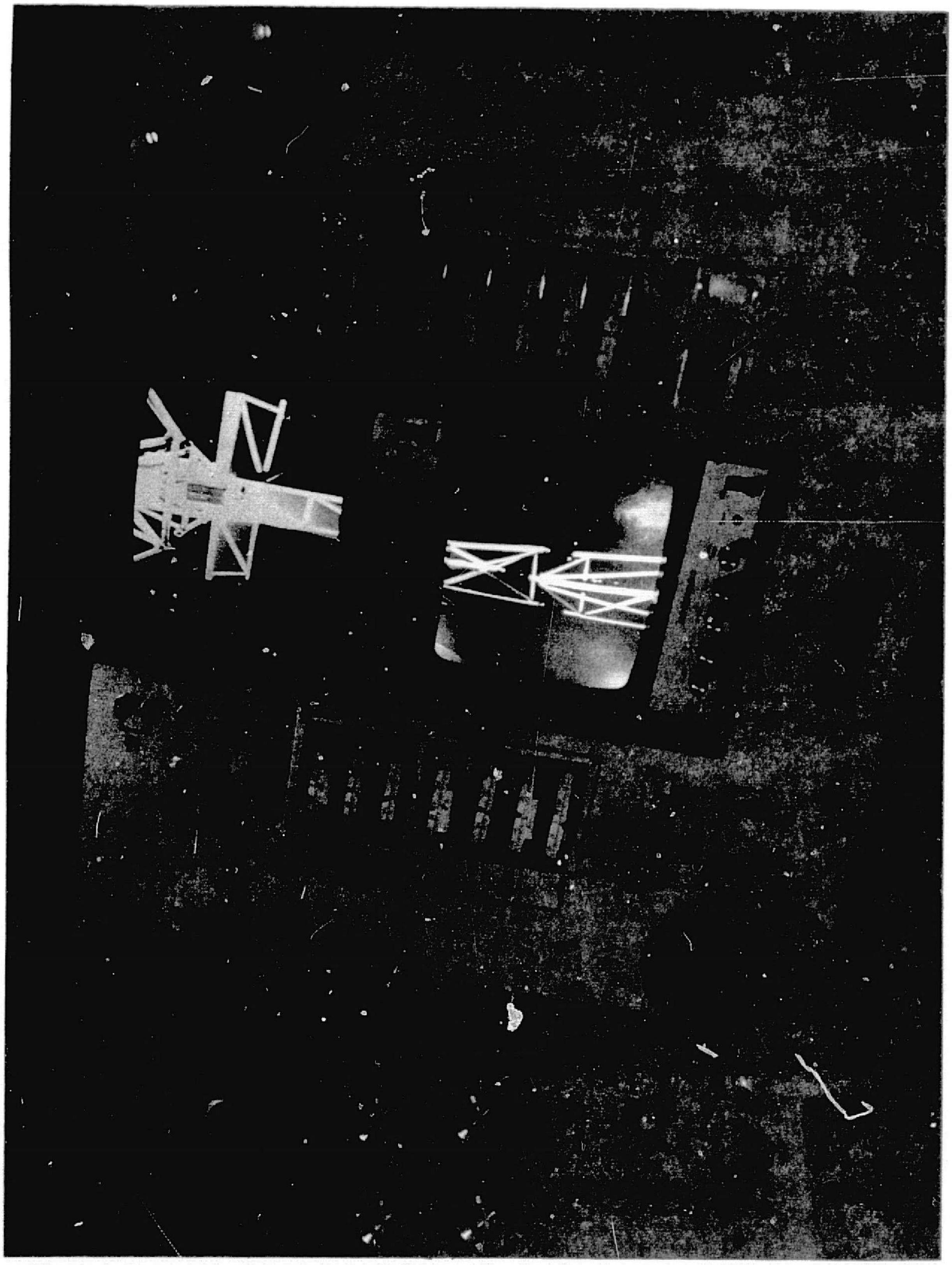

Figure IIF-32 MPTS Assembly Simulation, Task 2 - Beam Initial Alignment, Video View

The female or receptical beam tube ends were flaired slightly to provide a funnel effect. The male tube ends, on the beam segment attached to the SMA, had rounded tapered ends to facilitate insertion. These ends were painted black, with white tips, to increase the operator's ability to distinguish the tips from the surrounding structure.

The total average run time ran approximately 2.8 minutes. Seventy-five to 80% of this total time was used for the final alignment phase, which averaged approximately 2 minutes. We anticipated significant problems in aligning these beam ends prior to the actual simulation. However, the task was not difficult if the capability to release the manipulator wrist torques is available to the operator. The operators flew the two bottom tips into the receptacles using the end effector camera. This alignment required positional accuracies of $\pm 1/16$ in. The top tip was monitored through the right shoulder camera. As the pins were inserted, a small misalignment would prevent one or two of the pins from fully seating. By turning the wrist torques to zero, an (X) translation command would backdrive the manipulator wrist gimbals, which in turn completely inserted all three pins. The final alignment phase is shown in Figure IIF-33 and 34. The operators video view of this phase is shown in Figures II-32 and 33.

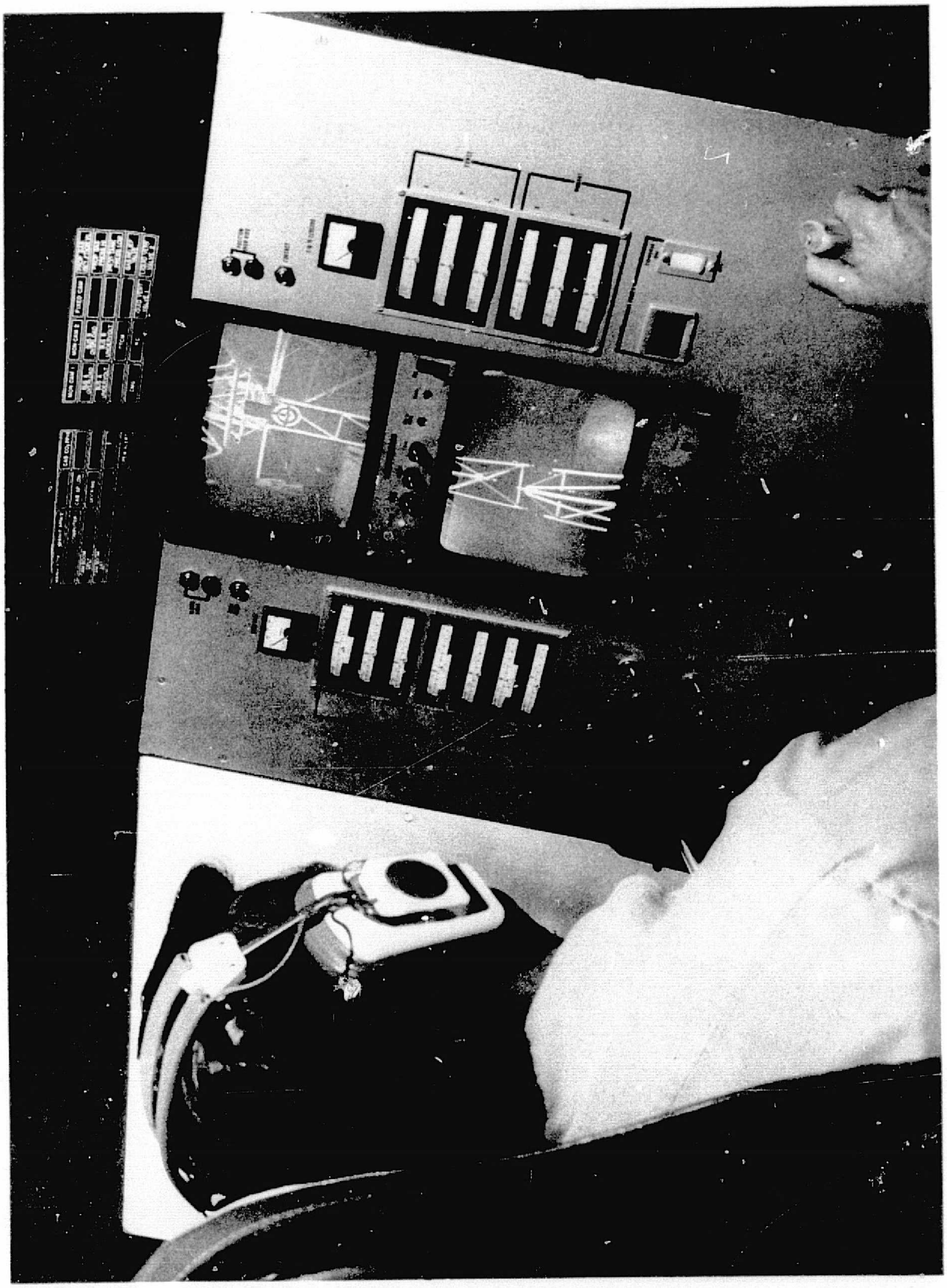

Figure IIF-33 MPTS Assembly Simulation, Task 2 - Beam Final Alignment, Video View

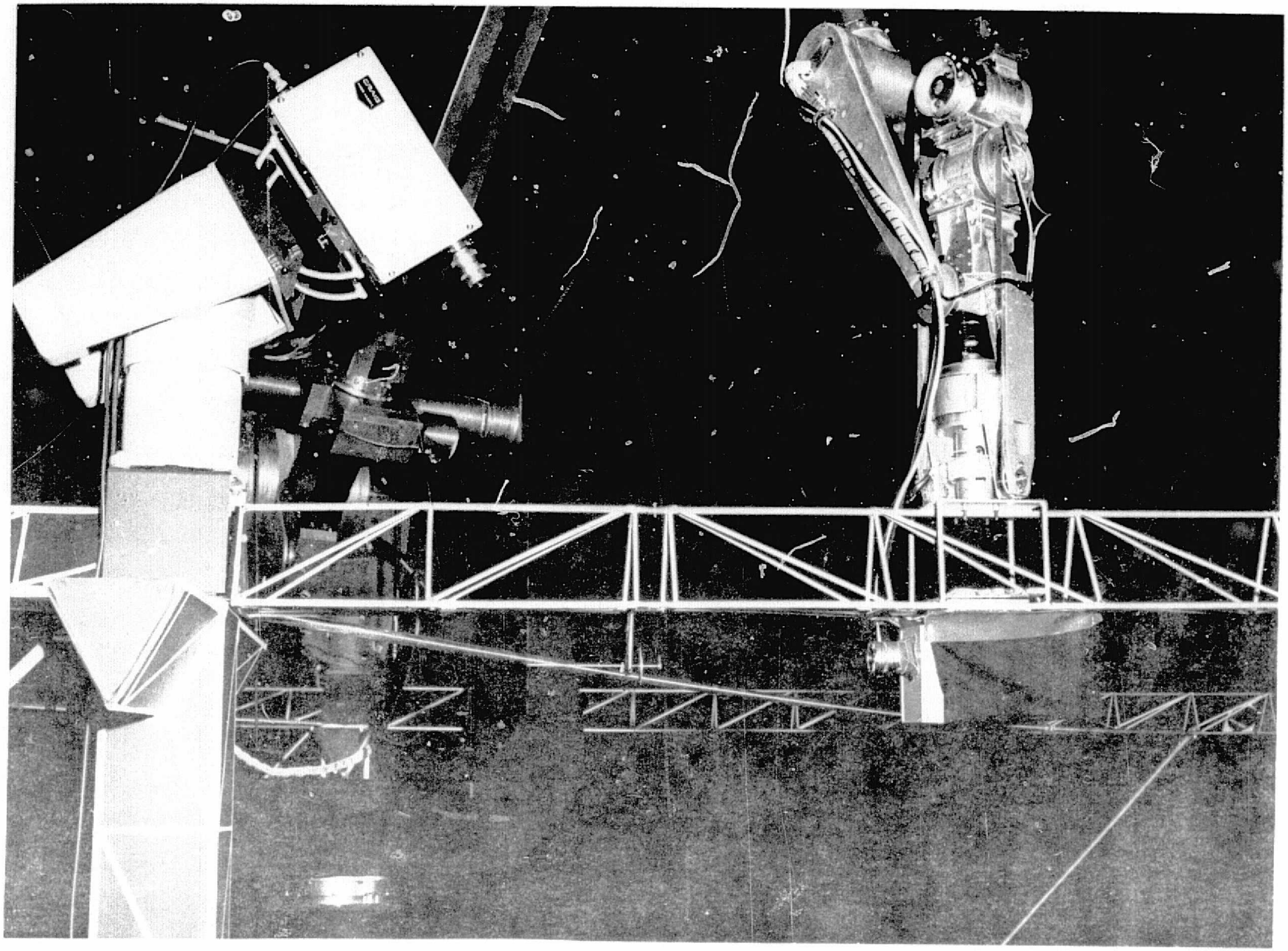

Figure IIF-34 MPTS Assembly Simulation, Task 2 - Beam Attached

6. Results and Recommendations

a. Test Subject - Three MMC operators (engineers) were used for the simulation. These test subjects have had considerable prior experience in the operation of the SMA and use of rate-type control systems. The familiarity of these operators with the equipment allowed expeditious selection of the optimum simulation configuration while minimizing training time.

In each phase of the simulation, the task was repeated until one of the operators reached an adequate point on the learning curve, and the test setup--i.e., camera locations, attach mechanisms, SMA control systems characteristics--were satisfactorily refined. The development of the final test setups required approximately 20 runs per major task. After a satisfactory setup was achieved, the other two operators learned the final task rapidly, requiring only 4 to 6 runs before their times stabilized. The run times averaged 2 to 4 min. for each of the three assembly tasks. Initial run times were from 8 to 10 min. The translation phases were readily learned. The final beam alignment and/or attachment was more demanding on the operator.

b. Simulation Results and Discussion Summary - Tables IIF-4 through IIF-12 show performance data summaries for each of the three operators, while conducting the three simulated space assembly operations. Each of these operations is broken into subtasks with associated performance data shown. As can be seen on the tables, the run times were fairly consistent between operators S and J. Operator R did average longer times and greater standard deviations; however, he had the least total experience on the SMA. All of the numerical data, shown in these summary tables, are averaged for 10 runs. A total of 90 data runs were conducted. The test crew carefully monitored each run and all inadvertant beam contacts were recorded. There were a total of five contacts, none of these were more than minor bumps and none caused any damage. Operator S contacted the double - ended MPTS beam once. Operators J and S each inadvertantly contacted the RAT beam twice during the final attachment phase. There were no inadvertant contacts during the single-ended MPTS beam assembly.

The operators were allowed to select the reference (control axis) video camera they wished as the task progressed. Each operator consistently chose the same control and reference cameras and the automatic wrist attitude hold mode without instruction. Similar translation and attitude velocities were also selected, without instruction. These velocities appear to be quite low, averaging generally below 1 ft/sec translation and 4-9 deg/sec attitude change. However, the task is physically scaled and ratioed 1:4 for the RAT beam and 1:5 for the MPTS beams. Therefore, these average and maximum translational velocities must be multiplied by 4 for the RAT beam data and 5 for the MPTS data to derive the tip velocities expected for the operational case. This would establish major translation tip velocities for the operational system at 2.5 to 5.0 ft/sec, which we

Table IIF-4 Simulation-MPTS Single-Ended Beam Installation (Task 2 - Subject S)

Task Breakdown	Task Times: Average (sec)	Task Times: % Total	Auto Attitude Hold Mode	Control Axis Camera	Pilot Selected Max Translation Tip Vel. (ft/sec)	Pilot Selected Max Rotational Vel. (deg/sec)	Wrist Torque	Alignment Aids	Operators Remarks
o Translate beam segment to installation site			Range	Left Shoulder	0.55 (0.58[1]/.054[2])	10.3 (9.7[1]/7.2[2])	On	None	Translation task minimized since it was identical to double-ended beam
o Maneuver into position	45.0	25%	"	"	"	"	"	Beam ends	Coordinated attitude & position maneuver
o Coarse align			"	"	"	"	"	"	Wrist camera used for position reference
o Close & fine align			Full	Wrist	0.15	2.1	"	"	Right shoulder camera used for reference
o Contact & insert alignment pins	128.0	74%	"	"	"	"	Zero wrist torque	"	Minimum motion of <1/8 in. required for this task
o Verify insertion			"	"	-	-	-	"	Right shoulder camera provided adequate view
Total Time Standard Deviation	173 sec 62 sec	100%							

NOTE: Except where specified, all numerical data is average for 10 runs.

1. Recorded maximum velocity.

2. Average recorded maximum velocity.

Table IIF-5 Simulation-MPTS Single-Ended Beam Installation (Task 2 - Subject R)

Task Breadsown	Task Times Average (sec)	Task Times % Total	Auto Attitude Hold Mode	Control Axis Camera	Pilot Selected Max Translation Tip Vel. (ft/sec)	Pilot Selected Max Rotational Vel. (deg/sec)	Wrist Torque	Alignment Aids	Operators Remarks
o Translate beam segment to installation site			Range	Left Shoulder	0.92 (1.10[1]/0.77)	8.7 (8.6[1]/7.2[2])	Full on		
o Maneuver into position	54.3	26%	"	"	"	"	"		
o Course align			"	"	"	"	"	Stand-off cross & beam ends	Had trouble determining Z-axis alignment
o Close & fine align			"	"	0.11	3.9	"	"	Same as above, also need high contrast tips on ends of beams
o Contact & insert alignment pins	154.7	74%	"	"	"	"	Full on-off	Beam ends	Need high contrast cues
o Verify insertion			"	"	"	"	-	-	Need high contrast cues
Total Time Standard Deviation	209 sec 38 sec	100%							

NOTE: Except where specified, all numerical data is average for 10 runs.

1. Recorded maximum velocity.

2. Average recorded maximum velocity.

Table IIF-6 Simulation-MPTS Single-Ended Beam Installation (Task 2 - Subject J)

Task Breakdown	Task Times: Average (sec)	Task Times: % Total	Auto Attitude Hold Mode	Control Axis Camera	Pilot Selected Max Translation Tip Vel. (ft/sec)	Pilot Selected Max Rotational Vel. (deg/sec)	Wrist Torque	Alignment Aids	Operators Remarks
o Translate beam segment to installation site	26	19%	Range		0.73 (0.75[1]/0.70[2])	11.0 (10.9[1]/8.7[2])	On	None	
o Maneuver into position			"		"	"	"	Beam ends	
o Course align			"		"	"	"	"	Standoff cross used initially, then abandoned.
o Close & fine align	111.0	81%	Full		0.14	3.6	"	"	Beam ends adequate for alignment aid.
o Contact & insert alignment pins			"		"	"	Zero wrist torque	"	Beam pitch had to be monitored & controlled from left shoulder camera. Better pitch cue needed.
o Verify insertion					"	"	-	"	
Total Time	137 sec	100%							
Standard Deviation	13 sec								

NOTE: Except where specified, all numerical data is average for 10 runs.

1. Recorded maximum velocity.

2. Average recorded maximum velocity.

Table IIF-7 Simulation-MPTS Double-Ended Beam Installation (Task 1 - Subject S)

Task Breakdown	Task Times Average (sec)	Task Times % Total	Auto Attitude Hold Mode	Control Axis Camera	Pilot Selected Max Translation Tip Vel. (ft/sec)	Pilot Selected Max Rotational Vel. (deg/sec)	Wrist Torque	Alignment Aids	Operators Remarks
o Extract beam from pallet			Range	Left Shoulder	0.58 (0.72[1]/0.58[2])	7.0 (6.9[1]/6.1[2])	On	None	Beam was totally deployed & attached to manipulator
o Translate beam to installation site			"	"	"	"	"	"	Range hawk allowed operator to yaw manipulator shoulder for major translation
o Maneuver into position	120	58%	"	"	"	"	"	"	Elbow & shoulder angle readout desirable to aid in determining manipulator extension in X direction
o Course align			"	"	"	"	"	Existing beam ends	Operator uses both shoulder cameras to view opposite beam ends
o Close & fine align	87	42%	Full	Right wrist	0.17	2.0	"	Alignment cross-hairs	Alignment aids on beam ends are required

Table IIF-7 (Concluded)

Task Breakdown	Task Times: Average (sec)	Task Times: % Total	Auto Attitude Hold Mode	Control Axis Camera	Pilot Selected Max Translation Tip Vel. (ft/sec)	Pilot Selected Max Rotational Vel. (deg/sec)	Wrist Torque	Alignment Aids	Operators Remarks
o Contact & verify alignment	↓	↓	Full	Right wrist	0.17	2.0	On	Alignment cross-hairs	Alignments within $\pm 1/16$ in. were possible
Total Time Standard Deviation	207 sec 33 sec	100%							

NOTE: Except where specified, all numerical data is average for 10 runs.

1. Recorded maximum velocity.

2. Average recorded maximum velocity.

Table IIF-8 Simulation-MPTS Double-Ended Beam Installation (Task 1 - Subject R)

Task Breakdown	Task Times: Average (sec)	Task Times: % Total	Auto Attitude Hold Mode	Control Axis Camera	Pilot Selected Max Translation Tip Vel. (ft/sec)	Pilot Selected Max Rotational Vel. (deg/sec)	Wrist Torque	Alignment Aids	Operators Remarks
o Extract beam from pallet	148.8	48%	Range	Left Shoulder	0.83 (1.22[1]/0.80[2])	10.0 (9.7[1]/6.9[2])	Full on	None	Operationally would need two cameras or larger field-of-view
o Translate beam to installation site			"	"	"	"	"	"	" " "
o Maneuver into position			"	"	"	"	"	"	Operationally would need wrist rotations in camera coordinates
o Course align			"	"	"	"	"	Existing beam ends	Further research needed on axis polarity. Pilot often used trial & error, esp. X-axis
o Close & fine align	161.2	52%	Full	Right wrist	0.09	2.6	"	Alignment cross-hairs	Same as above
o Contact & verify alignment			"	"	"	"	"	Beams & cross-hairs	
Total Time	310 sec	100%							
Standard Deviation	48 sec								

NOTE: Except where specified, all numerical data is average for 10 runs.

1. Recorded maximum velocity.
2. Average recorded maximum velocity.

Table IIF-9 Simulation-MPTS Double-Ended Beam Installation (Task 1 - Subject J)

Task Breakdown	Task Times: Average (sec)	Task Times: % Total	Auto Attitude Hold Mode	Control Axis Camera	Pilot Selected Max Translation Tip Vel. (ft/sec)	Pilot Selected Max Rotational Vel. (deg/sec)	Wrist Torque	Alignment Aids	Operators Remarks
o Extract beam from pallet			Range	Left shoulder	0.83 ($0.98^1/0.76^2$)	10.0 ($9.4^1/7.6^2$)	Full on	None	Zoom lens on shoulder cameras desirable
o Translate beam to installation site	102	49%	"	"	"	"	"	"	Auto track camera capability desirable
o Maneuver into position			"	"	"	"	"	"	Manipulator range display needed for extension
o Course align			"	"	"	"	"	Existing beam ends	Two shoulder cameras needed
o Close & fine align	106	51%	Full	Right wrist	0.13	2.9	"	Alignment cross-hairs	Considerable trial and error
o Contact & verify alignment			"	"	"	"	"	" "	Present system adequate but could be improved
Total Time Standard Deviation	208 sec 56 sec	100%							

NOTE: Except where specified, all numerical data is average for 10 runs.

1. Recorded maximum velocity.
2. Average recorded maximum velocity.

Table IIF-10 Simulation-RAT Beam Installation (Subject S)

Task Breakdown	Task Times Average (sec)	Task Times % Total	Auto Attitude Hold Mode	Control Axis Camera	Pilot Selected Max Translation Tip Vel. (ft/sec)	Pilot Selected Max Rotational Vel. (deg/sec)	Joint Torque	Alignment Aids	Operators Remarks
o Extract beam from cargo bay	35.6	26%	Full	Left Shoulder	0.46 ($0.46^1/0.38^2$)	4.0 ($4.0^1/3.3^2$)	All on	None	Internal cargo bay cameras may be required
o Translate to core			"	"	"	"	"	"	Shuttle tail presents a significant obstical for 55-ft long beam
o Position & align at attach point	73.9	54%	"	End Effector	0.14	"	"	Stand-off cross	Supplemental alignment aid required. Attachment device not adequate for visual alignment
o Critical align			"	"	"	"	"	Attachment mech.	Requires minimum motion & stabilization of <1/8 in.
o Contact lower attach pins & position in groves			"	"	"	"	"	"	Attachment device must have non-glare surface with high contrast pins
o Rotate upper attach pins & contact	27.4	20%	"	"	"	"	Shoulder off	"	Manipulator shoulder torque release required

Table IIF-10 (Concluded)

Task Breakdown	Task Times		Auto Attitude Hold Mode	Control Axis Camera	Pilot Selected Max Translation Tip Vel. (ft/sec)	Pilot Selected Max Rotational Vel. (deg/sec)	Joint Torque	Alignment Aids	Operators Remarks
	Average (sec)	% Total							
o Verify attachment			-	-	0.14	4.0 (4.0[1]/3.3[2])	Shoulder off	Attachment mech	Visual or electrical verification required
Total Time Standard Deviation	137 sec 33 sec								

NOTE: Except where specified, all numerical data is average for 10 runs.

1. Recorded maximum velocity.

2. Average recorded maximum velocity.

Table IIF-11 Simulation-RAT Beam Installation (Subject R)

Task Breakdown	Task Times: Average (sec)	Task Times: % Total	Auto Attitude Hold Mode	Control Axis Camera	Pilot Selected Max Translation Tip Vel. (ft/sec)	Pilot Selected Max Rotational Vel. (deg/sec)	Joint Torque	Alignment Aids	Operators Remarks
o Extract beam from cargo bay	52.7	34%	Full	Left Shoulder	0.36 (0.27[1]/0.24[2])	4.6 (3.2[1]/2.7[2])	Full on		Operationally need two cameras or larger field-of-view
o Translate to core			"	"	"	"	"		Need two cameras or larger field-of-view
o Position & align at attach point	80.6	52%	"	End effector	0.23	"	"		Need high contrast between alignment guide & mating pins
o Critical align			"	"	"	4.4	"		Same as above also need pure rotations not coupled
o Contact lower attach pins & position in groves			"	"	"	"	"		Same as above
o Rotate upper attach pins & contact	21.7	14%	"	"	"	"	Shoulder off		Turned off shoulder torque very easy to drive whole arm with wrist pitch torque

Table IIF-11 (Concluded).

Task Breakdown	Task Times Average (sec)	Task Times % Total	Auto Attitude Hold Mode	Control Axis Camera	Pilot Selected Max Translation Tip Vel. (ft/sec)	Pilot Selected Max Rotational Vel. (deg/sec)	Joint Torque	Alignment Aids	Operators Remarks
o Verify attachment	↓	↓	Full	End Effector	-	4.4	Shoulder off		Need high contrast visual aids
Total Time	155 sec								
Standard Deviation	60 sec								

NOTE: Except where specified, all numerical data is average for 10 runs.

1. Recorded maximum velocity.

2. Average recorded maximum velocity.

Table IIF-12 Simulation-RAT Beam Installation (Subject J)

Task Breakdown	Task Times: Average (sec)	Task Times: % Total	Auto Attitude Hold Mode	Control Axis Camera	Pilot Selected Max Translation Tip Vel. (ft/sec)	Pilot Selected Max Rotational Vel. (deg/sec)	Joint Torque	Alignment Aids	Operators Remarks
o Extract beam from cargo bay	39.1	34%	Full	Left Shoulder	0.40 (0.44[1]/0.32[2])	2.6 (2.5[1]/2.1[2])	All on	None	Additional camera views may be required
o Translate to core			"	"	"	"	"	"	Same as above
o Position & align to attach point			"	End effector	0.25	"	"	Stand-off cross	Alignment aid required
o Critical align	75.9	66%	"	"	"	"	"	Attachment mech.	"Y" axis translation much more critical than others
o Contact lower attach pins & position in groves			"	"	"	"	"	"	High contrast target required
o Rotate upper attach pins & contact	TOTAL 115 sec Std.Dev. 31 sec	NOT RUN							
o Verify attachment									

NOTE: Except where specified, all numerical data is average for 10 runs.

1. Recorded maximum velocity.
2. Average recorded maximum velocity.

feel to be quite reasonable. The manipulator shoulder and wrist angular rates will remain constant. The task times can also be expected to remain constant between the simulations and the operational case.

Both the MPTS single-ended beam and the RAT beam installation required positioning after initial contact with its mating surface. The RAT beam connection required several motions to complete the latch sequence. Our initial attempts to make these maneuvers with full wrist and shoulder torques were unsuccessful. The operator had difficulty determining which corrective action to take when contact forces built up inadvertantly while attempting an attachment. We found we could not make the RAT beam attachment without setting the manipulator shoulder torques to zero and commanding a wrist pitch. This pitched the beam attachment into place while backdriving the shoulder into the desired orientation. This feature was desirable, but not mandatory on the MPTS single-ended beam attachment.

As the simulation data was reduced an interesting phenomena was noted. Each operator tended to make minimum impulse commands with the controllers as the beam alignments became critical. These impulses were less than one second in duration and were made in discrete axis only. No coordinated motions were commanded when the operator used the impulse command mode. During the final alignment of the double-ended MPTS beam, for example, 50% of the commands were a minimum impulse type. This control characteristic should be considered when developing the requirements for an operational system since the controllers used in this simulation are not optimized for the impulse mode. An operational system could have the capability to switch from the standard proportional rate type controller characteristics to controller characteristics for an impulse mode during final alignments.

c. Utilization of Manipulator Characteristics - The orbital assembly system made use of the 12-ft Slave Manipulator Arm (SMA) to perform size-scaled tasks. Because the SMA is a general purpose tool, its capabilities are generally not completely utilized in any given task. A special purpose manipulator, such as the Mobile Assembler for example, may not require the 30 degree-per-second maximum joint angle rates of the SMA.

Table IIF-13 shows the one-time maximum and average maximum joint angle rates for comparison with the maximum rates of which the SMA is capable. As can be seen in the table the actual rates rarely exceeded 30 percent of the design maximum and only once was greater than 50 percent. The three Martin Marietta pilots therefore did not at any time utilize the design maximum rates. Although the sampling is small with only three participants, the rates used were so low that there is a strong indication that a maximum rate of 10 degrees-per-second would be sufficient as a design maximum for the space counterpart--perhaps as low as 6 or 7 degrees-per second for the two shoulder gimbals.

Table IIF-13 Comparison of Maximum Joint Angular Rates

TASK	PILOT	GIMBAL RATES DEGREES/SECOND																	
		SHOULDER YAW			SHOULDER PITCH			ELBOW PITCH			WRIST ROLL			WRIST PITCH			WRIST YAW		
		1-TIME MAX	AVG MAX	DESIGN MAX	1-TIME MAX	AVG MAX	DESIGN MAX	1-TIME MAX	AVG MAX	DESIGN MAX	1-TIME MAX	AVG MAX	DESIGN MAX	1-TIME MAX	AVG MAX	DESIGN MAX	1-TIME MAX	AVG MAX	DESIGN MAX
Double-Ended	J	6.9	5.3	30	7.2	5.2	30	12.9	8.8	30	8.3	7.1	30	9.9	6.2	30	8.9	7.6	30
Beam	R	9.2	6.6	"	10.7	5.6	"	18.3	9.2	"	10	6.8	"	6	2.1	"	9.7	6.9	"
Attachment	S	4.6	3.9	"	10.3	5.7	"	16	9.3	"	9.7	6.1	"	8.0	2.7	"	6.9	5.9	"
Single-Ended	J	6.0	4.1	"	6.1	3.6	"	10.9	6.7	"	9.7	8.3	"	7.3	2.9	"	9.7	8.7	"
Beam	R	8.1	5.0	"	5.4	4.2	"	8.6	5.3	"	7.2	6.9	"	6.3	3.2	"	8.6	7.2	"
Attachment	S	4.9	4.0	"	3.6	2.9	"	7.0	5.6	"	7.3	6.8	"	9.7	5.6	"	7.4	7.1	"
RAT	J	1.8	0.6	"	3.3	2.3	"	2.0	1.5	"	2.5	2.1	"	2.0	1.2	"	2.4	1.7	"
Beam	R	2.3	1.8	"	2.2	1.6	"	2.9	2.7	"	3.1	2.5	"	3.2	1.8	"	2.0	1.6	"
Attachment	S	4.2	2.8	"	4.1	3.4	"	5.8	4.0	"	4.0	3.3	"	2.2	1.5	"	3.9	2.6	

Table IIF-14 shows the range of gimbal angles used compared to the maximum allowable angles. Both singly-occurring maximums and typical maximum (avg. max) excursions are given. These figures are compiled for the double-ended beam and single-ended beam attachments separately as well as in combination since both are performed by the Mobile Assembler (MA). The third part of Table IIF-14 shows the overall range of gimbal angles needed to perform the two tasks simulated. To determine the complete MA angular criteria it will be necessary to define all the MA tasks including those necessary for self-repositioning.

The fourth part of Table IIF-14 gives the corresponding angular displacement maximums for the RAT beam assembly. This exercise is, of course, only one of many tasks for the Shuttle RMS and should not be considered a driving factor for the RMS design except that the RAT beam requirements be included within the RMS operational envelope.

Obviously, the angular displacement data is highly task dependent. It is less dependent than the rate data upon the individual operator. Thus, the figures in Table IIF-14 are averaged for the three operators.

In design of a manipulator, it is important to simulate physically or with a computer, the maximum excursions of each gimbal. As a minimum, the manipulator must encompass these maximum displacements. However, to accommodate unforeseen contingencies, the manipulator might best be designed with as much over-range as possible. To aid in avoiding hazards such as the base structure, simple microprocessor logic in the control system can define forbidden zones into which the manipulator may not move.

d. Conclusions - We have demonstrated that the proposed, in-space assembly technique, using a remotely controlled manipulator is feasible. We have shown that the beam assembly times are less than anticipated and that these times can be reduced through the use of preprogrammed translation control modes. A simplified proportional rate control system was successfully used. Not only was this control system found acceptable, but highly desirable. This demonstration de-emphasizes the need for a complicated manipulator control system such as used with a force feedback (bilateral) position controller. The following manipulator characteristics should be incorporated in the proposed mobile assembler system:

- Coordinated manipulator control motions are required for these in-space assembly tasks.
- Manipulator control axis alignment with the video system camera used for the prime visual feedback is mandatory.
- Manipulator shoulder and wrist torque output control is required at the operator's console.

Table IIF-14 Comparison of Maximum Joint Angle Displacements

	TASK	DIRECTION	GIMBAL ANGULAR DISPLACEMENT: DEGREES																	
			SHOULDER YAW			SHOULDER PITCH			ELBOW PITCH			WRIST ROLL			WRIST PITCH			WRIST YAW		
			1-TIME MAX	AVG MAX	DESIGN MAX	1-TIME MAX	AVG MAX	DESIGN MAX	1-TIME MAX	AVG MAX	DESIGN MAX	1-TIME MAX	AVG MAX	DESIGN MAX	1-TIME MAX	AVG MAX	DESIGN MAX	1-TIME MAX	AVG MAX	DESIGN MAX
MPTS	Double-Ended Beam Attachment	+	155	148	200	109	73	150	-27	-42	10	23	11	130	13	2	80	10	-2	200
		-	-9	-1	-200	-7	1	-75	-107	-82	-160	-89	-63	-130	-15	-4	-80	-107	-101	-200
	Single-Ended Beam Attachment	+	95	91	200	49	47	150	-65	-75	10	97	52	130	21	2	80	108	101	200
		-	41	50	-200	5	16	-75	-132	-124	-160	-71	-51	-130	-31	-8	-80	0	25	-200
	Both Above Tasks	+	155	148	200	109	73	150	-27	-42	10	97	52	130	21	2	80	108	101	200
		-	-9	-1	-200	-7	1	-75	-132	-124	-160	-89	-63	-130	-31	-8	-80	-107	-101	-200
RAT	Beam Attachment	+	2	-7	200	86	76	150	-61	-80	10	-46	-53	130	12	4	80	-60	-65	200
		-	-34	-31	-200	24	43	-75	-109	-95	-160	-97	-84	-130	-9	-5	-80	-100	-91	-200

- o A partial (range) and fully automatic manipulator wrist attitude hold modes are required.
- o Supplemental alignment aids, such as cross-hairs and standoff crosses are required for final beam positioning and alignment. The alignment aid technique used on the operational system should be standardized throughout the total assembly.

III. ASSEMBLY OF RADIO ASTRONOMY TELESCOPE

A. REQUIREMENTS (TASK 1)

This chapter addresses the orbital assembly of a radio astronomy telescope (RAT) which is 200 meters in diameter.

The 200-meter diameter structure was felt to be quite representative of the many medium-size rigid structures of the coming Shuttle era. Scientifically, this 200-meter diameter parabolic antenna will be used to detect RF sources in the celestial sphere in the 5 to 10 MHz band. It will be placed in an 8,000 n mi altitude circular orbit of 0 deg inclination. Figure IIIA-1 is a simple depiction of the satellite in an operational condition. It will be rigid because this represents a more realistic example of assembly techniques. Unique designs such as "LOFT" (Low Frequency Telescope) which feature very lightweight mesh that requires spinning for deployment and maintaining operational shape have their own peculiar problems and applications which are not general enough to warrant their study on this contract.

Because a baseline design was not available from any previous work, we generated a baseline configuration and design that was felt to be quite reasonable and straightforward and used this to evaluate various assembly techniques. The design will be described in Section B in terms of its relationship to scientific needs of the antenna and the constraints imposed by Shuttle, Tug and other operational conditions identified in this section. This will be followed in Section C by a detailed description of various assembly approaches, their advantages and disadvantages, and a preliminary comparison of them. We will discuss these approaches qualitatively and identify the need for higher performing support equipment and/or different and more radical thinking in terms of configuration design.

Many of the dimensional requirements impacting the design are directly related to the wavelengths of the band we are investigating. In this case, 30 meters to 60 meters are the wavelengths (λ) corresponding to 10 MHz and 5 MHz, respectively.

A major requirement is for the surface to maintain its parabolic shape within an allowable tolerance. Standard antenna

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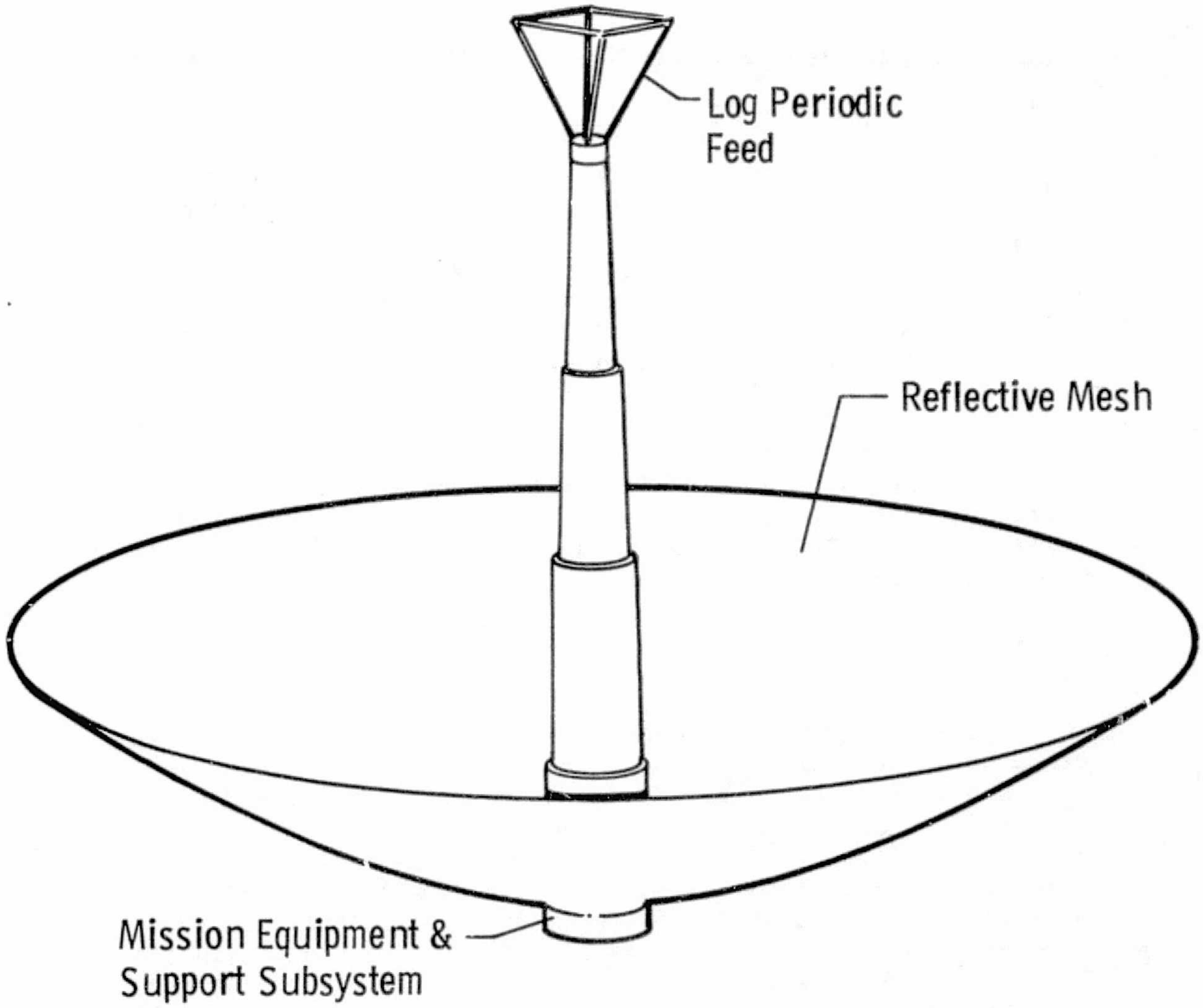

Figure IIIA-1 Radio Astronomy Telescope Concept

design practice dictates an allowable deviation of $\lambda/20$ (1.5 meters). The mast height which supports the antenna feed is equal to the focal length, chosen as 0.4D. This results in a length of 80 meters. The feed width should at least be as large as half the longest wavelength to be examined, hence 30 meters. Mast deflections of $\lambda/2$ in the axial direction (15 meter) and 1.5 m in the lateral direction are acceptable.

A significant effort was spent on the telescope antenna specifications. From this analysis it was determined that

eight ribs were needed to minimize the gain loss and from this the allowable surface deviation was determined. It was seen that leakage through the wire mesh was a contributor to back radiation (from earth). It was shown that 23 cm was the maximum grid spacing allowable to minimize this effect (10 cm was used in our design). Grid wire diameter is determined by skin depth requirements. Diameter should be at least 2 skin depths. At 5 MHz, one skin depth is 1 mil for copper. Two mils of copper grid wire was chosen.

A subsystem weight and power consumption analysis was performed and demonstrated little effect on the overall design (493 lbs and 34 watts). These estimates include RF cable runs, radio telescope feed to instrument package, instrumentation receiver and data processing, STDN S-band transponder (with ranging and command decoder at 2 watt RF power input), two low gain STDN antennas, and cabling to STDN antennas. All of the items chosen for this estimate are available or will be in the next few years.

Table IIIA-1 is a list of some of the salient baseline design characteristics.

Table IIIA-1 Baseline Design Characteristics

Frequency Band: 5 MHz (λ = 60 m) to 10 MHz (λ = 30 m)	
Resolution: 21.0 deg to 10.5 deg	
Allowable Surface Deviation: 1.5 meters	
Grid Spacing: 0.1 meter	
Pointing Accuracy: 1 deg	
Mast Height: 80 meters	
Feed Width: 30 meters	
Wire Gage: 2 mils	
Mast Deflections: 15 meters (axial); 1.5 meters (lateral) (Operationally allowable)	
Polarization: Dual Polarization Measurements	
Attitude Control System: Chemical thrusters	
Operational Requirements:	Attitude Hold for 2 hr (Pointing Mode)
	Slow slew for 180 deg/day (Scan Mode)

From these scientific requirements and the knowledge that the structure would have to be boosted with a Shuttle, a

configuration concept was developed whereby telescoping beams of 50-ft lengths could be compressed for launch and extended outward to form a spoke-like configuration as seen in Figure IIIA-2. A mesh material stretched over a wire support could be attached to the spoke armature and form the antenna while not exceeding unreasonable weight and volume. Subsystem impacts had to be ascertained and an analysis was performed to size the needed ACS jet thrusting level and location (this was based on scientific needs and disturbance constraints). The choice of jets was used to determine whether those loads at those locations would cause the surface smoothness requirement to be threatened. This was shown to be no problem. Propellant tanks, attitude sensors, electronics, docking systems, and other representative systems were chosen and evaluated as to their practical impositions to the basic design and were shown not to be limiting in any practical sense.

B. CONCEPTUAL DESIGNS (TASK 2)

This section describes the design effort of the study. Detail design is not an end-product of this study. It is required only to that depth necessary for analyzing the assembly process and maintenance requirements. However, insufficient design depth could lead to unrealistic assembly or maintenance approaches.

The main operational characteristics which affect the structural design are: 1) the allowable surface deviation, and 2) allowable mast deflection. These two things determine the number of support ribs required to maintain contours and stiffness requirements of the ribs and feed mast structures.

If these members are sized to accommodate orbital operational effects, they are relatively light and small. If the members are required to maintain a reasonable shape for thrusting by Tug, these members become much larger.

Obviously, from the standpoint of ease of assembly, it would be ideal to assemble the antenna complete in low earth orbit (LEO) where man and Shuttle will be normally operating. After assembly and checkout, the antenna would be boosted to high earth orbit (HEO). This is possible providing the antenna does not exceed a total weight of about 28,000 pounds, the

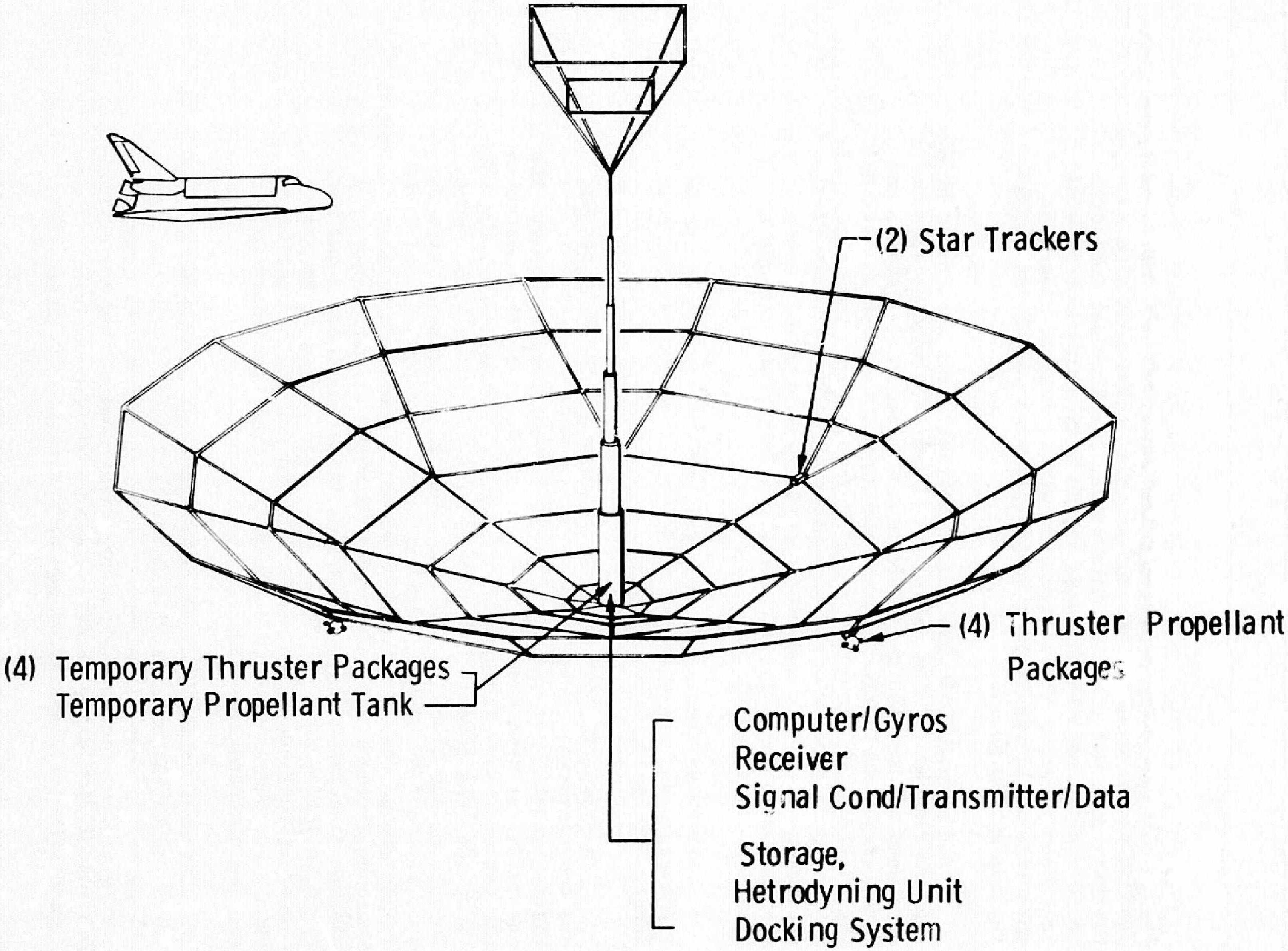

Figure IIIA-2 Radio Astronomy Telescope Assembly

approximate tug boost capability for the 8,000 mile orbit. The loading on the deployed antenna in this flight condition is slightly over 0.4 g's. At this value, the 320-feet long ribs would require beam thicknesses at the body attachment several feet deep to bring rib deflections within reason. Beams of this size would result in a total telescope weight far exceeding the 28,000 pounds.

For purposes of this study, only aluminum structures have been considered for the telescope. Even if composite structures were used to decrease the deflection to acceptable levels, the total weight would still exceed the 28,000 lbs. It was assumed that deflections exceeding about 50 inches would be unacceptable as damage to the net would probably occur. Operational surface deviations are not a consideration in this problem. Dynamic problems with such flexible structures may pose problems at boost cutoff as well.

All these conditions dictate that the structure must be lightweight and at least partially folded to shorten the beam length during the boost period.

In evaluating the problem of how small the beam can be, the consideration of net storage space becomes important. Figure IIIB-1 shows the net storage concept for all beams. The net reflector area over beams 1, 2, and 3, is small compared to the net area covering beams 4, 5, and 6. Consequently, the outer net area governs the minimum size for beams 4, 5, and 6. Nets D, E, and F are on rollers. The rolled diameter is a result of 120 feet of net, .005 in. thick. Nets G, H, and J are folded around the outside of these beams due to their non-rectangular shape when deployed. With the establishment of a minimum size for beams 4, 5, and 6, beams 1, 2, and 3 can be sized to accommodate the inner net. The beams also become larger as they are nearer the supported end of the cantilever beam. For purposes of this study, it is not necessary to pursue the structural analysis further.

Because of the low frequency and large wavelength characteristics of the antenna, 1.5 meter (4.9 feet) deviation from the parabolic shape is allowed. Rib type structures generate a series of flat sections which may be at nominal contour at the rib and maximum error occurring midway between the ribs. (See Figure IIIB-2.) This error can be substantially larger than

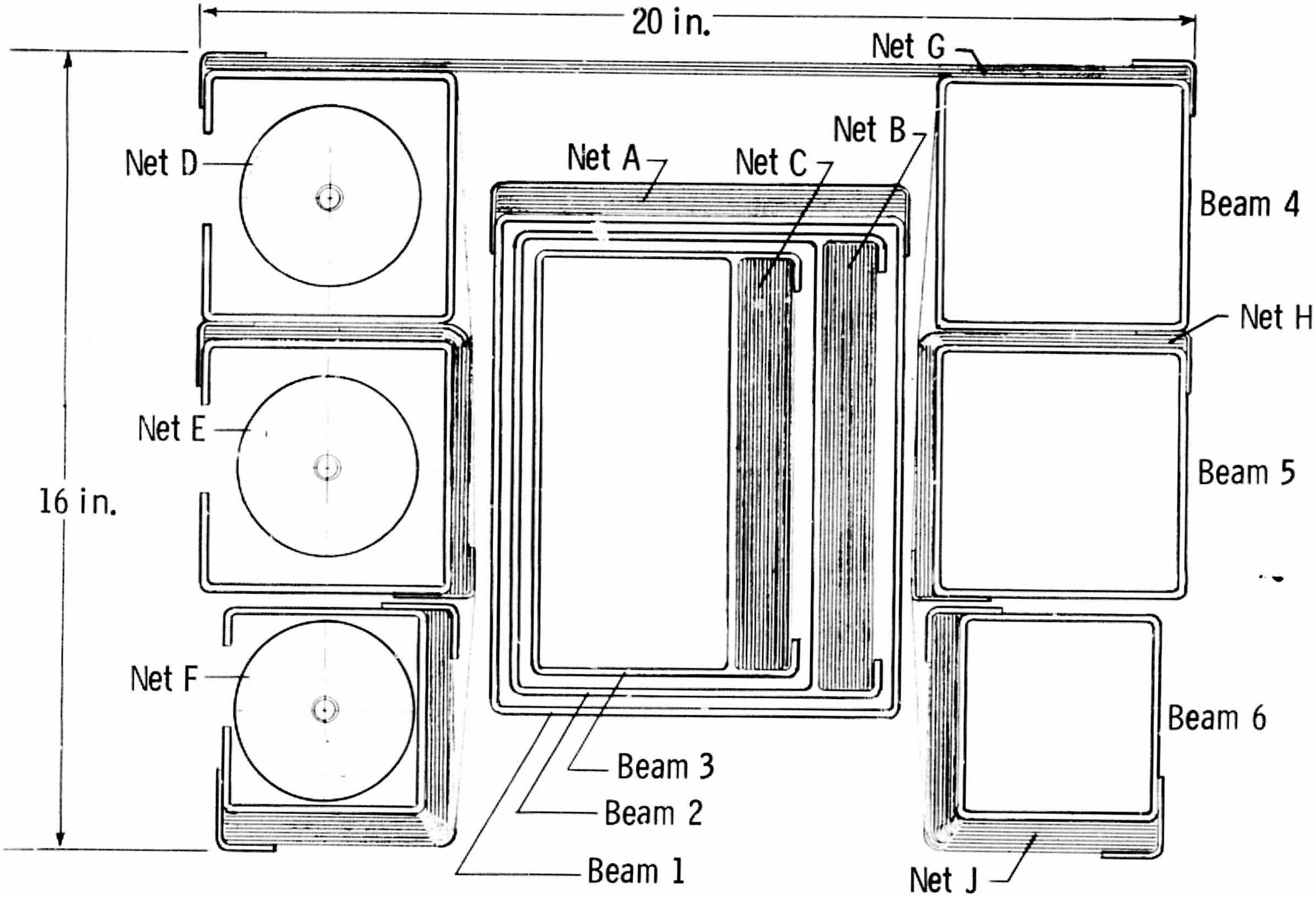

Figure IIIB-1 Beam Package Cross Section

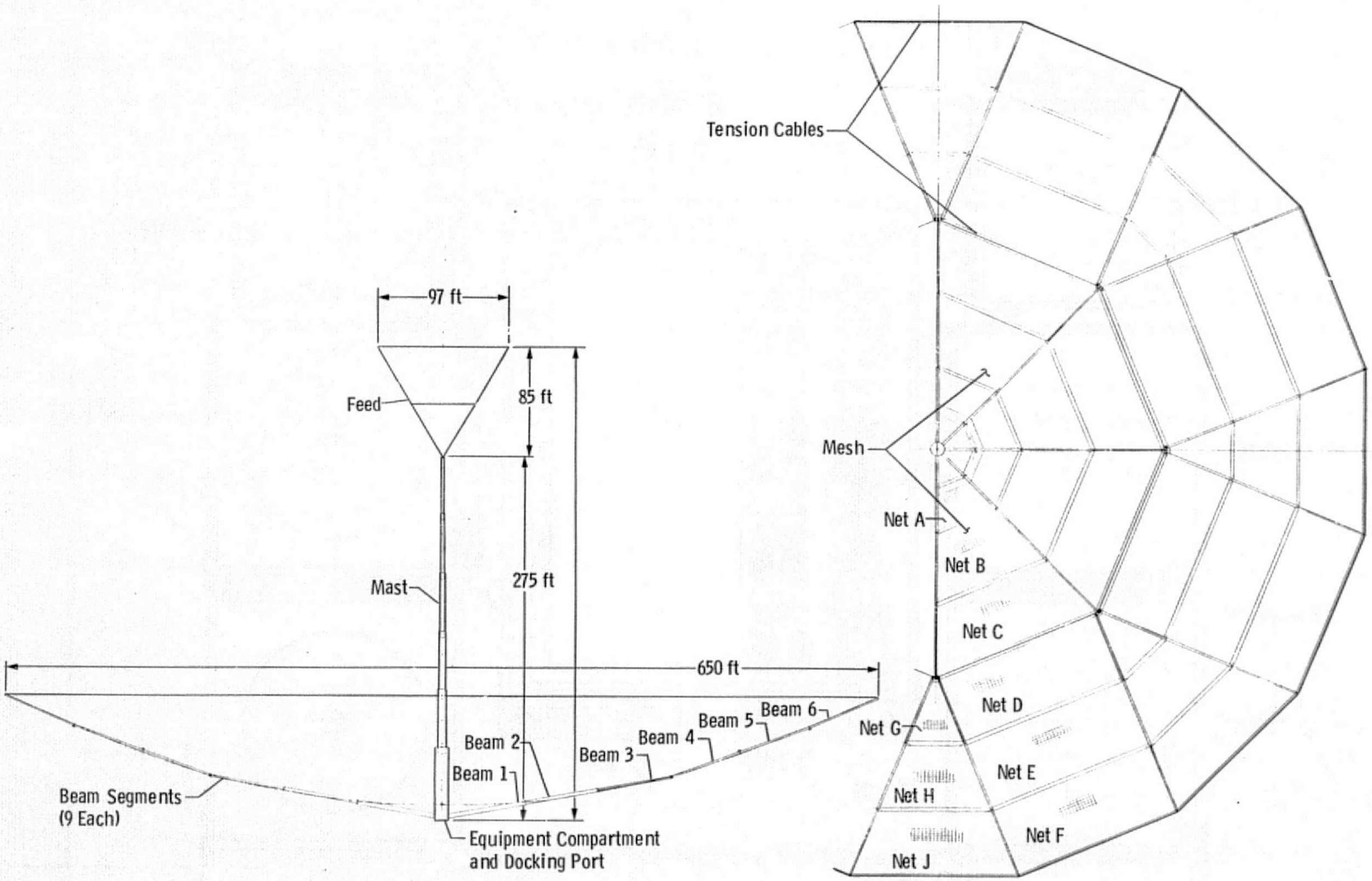

Figure IIIB-2 200-Meter Radio Astronomy Telescope

the allowable surface deviation as it occurs in only eight places and diminishes as the radius decreases. The baseline design incorporates diverging beams at the midpoint thus producing 16 outer beams. With the outer circumference divided 16 times rather than 8, the midpoint error drops from 11 ft to 3 ft. This provides for considerable antenna gain improvement over the 8 beam design. The beam sections of 55-foot length need not be contoured as the surface deviation over 55 feet on a 650-ft diameter parabola is very minimal.

Total deviation from nominal contour is affected also by manufacturing tolerances and forces causing beam and net deflections. The major contributor to beam deflection is the small thruster package located at the midpoint of four of the eight beams. By locating them approximately 160 ft from the antenna centerline, large torques are applied with small thruster pulses and low gas usage. If the thruster is in the range of one pound thrust, the beam deflection is small.

The Radio Astronomy Telescope was divided into three main components.

1) Central mast/feed structure;

2) Contour beam assemblies;

3) Reflector net.

The study revealed each of the above components can be designed very lightweight resulting in a low packaging density for Shuttle cargo bay transport. This particular fact will obviously affect any large structure requiring assembly in earth orbit. For instance, any beam of high inertia-to-weight ratios will have low density. In this case it was decided that the only way to increase packaging density while maintaining efficient structural design was to telescope and fold where possible and utilize remaining open spaces for net stowage. This is the philosophy of the beam design shown in Figure IIIB-1. Unfortunately, as you fold and telescope parts, the complexity increases. Moving parts with drive mechanisms become necessary in many cases where large masses are deployed into place.

The length of the cargo bay limits the length of any member to under 60 feet. Since the length of one rib is approximately 325 feet, the beam folded in six sections is about 54 feet long.

The central mast/feed structure presents the same problems as the contour beams. The mast and feed must be folded and telescoped so that reasonable cargo bay storage can be achieved.

Net design is based on RF considerations. Two mil diameter copper wire spaced 4 in. apart is the main requirement for proper reflectivity. Deployment of a woven net of these dimensions would most certainly result in a tangled mess that probably would not deploy regardless of what method of stowage is used. The addition of a 1/2-mil layer of mylar over each side of the copper grid forms a stable sheet which can be folded, rolled and manipulated without fear of tangling. The mylar becomes a major part of the net weight, but compared to the total antenna weight, it is not unusually high. If 0.004-in. wire is used in the grid and two 1/2-mil sheets are bonded in, the total thickness becomes 0.005 in. and many feet can be rolled up and stored within a beam. The distances between the beams are as much as 120 feet, so the roll diameter does become significant.

The design as shown in this report shows eight mechanically-deployed net-support ribs. These ribs require considerable mechanical equipment to effect deployment such as electromechanical drives, cables, rollers, etc. If the means are available in orbit to attach beam segments together, then the deployment mechanisms would be replaced by more simple attach hardware. The same is true for the feed mast structural assembly and the feed. On the other hand, telescoping structure is an excellent way to increase packaging densities. The 200-meter radio astronomy telescope can be transported to low earth orbit in one Shuttle launch in the telescoped and folded configuration shown here. If this design was altered to a single member configuration, at least two and probably three Shuttle flights would be required just to move the construction materials to LEO. These questions constitute one of the most important basis of this study, and will be pursued throughout the remainder of the study for all structures.

To summarize, the structural considerations and transport constraints imposed by Shuttle resulted in the antenna design described here. Figure IIIB-3 shows two rib sections attached to the central can. Figure IIIB-4 shows the net layout over

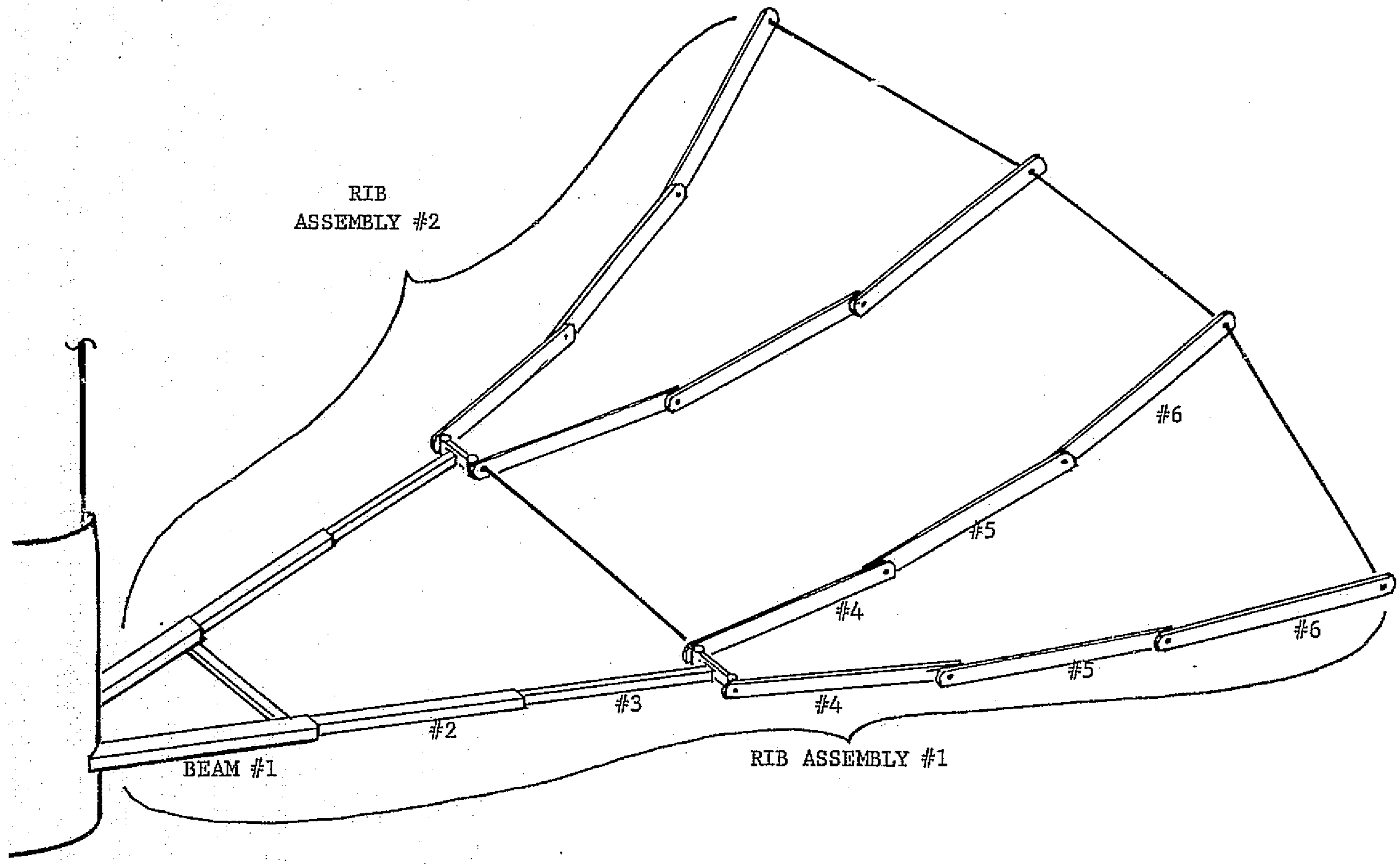

Figure IIIB-3 Radio Astronomy Telescope Beam Assembly

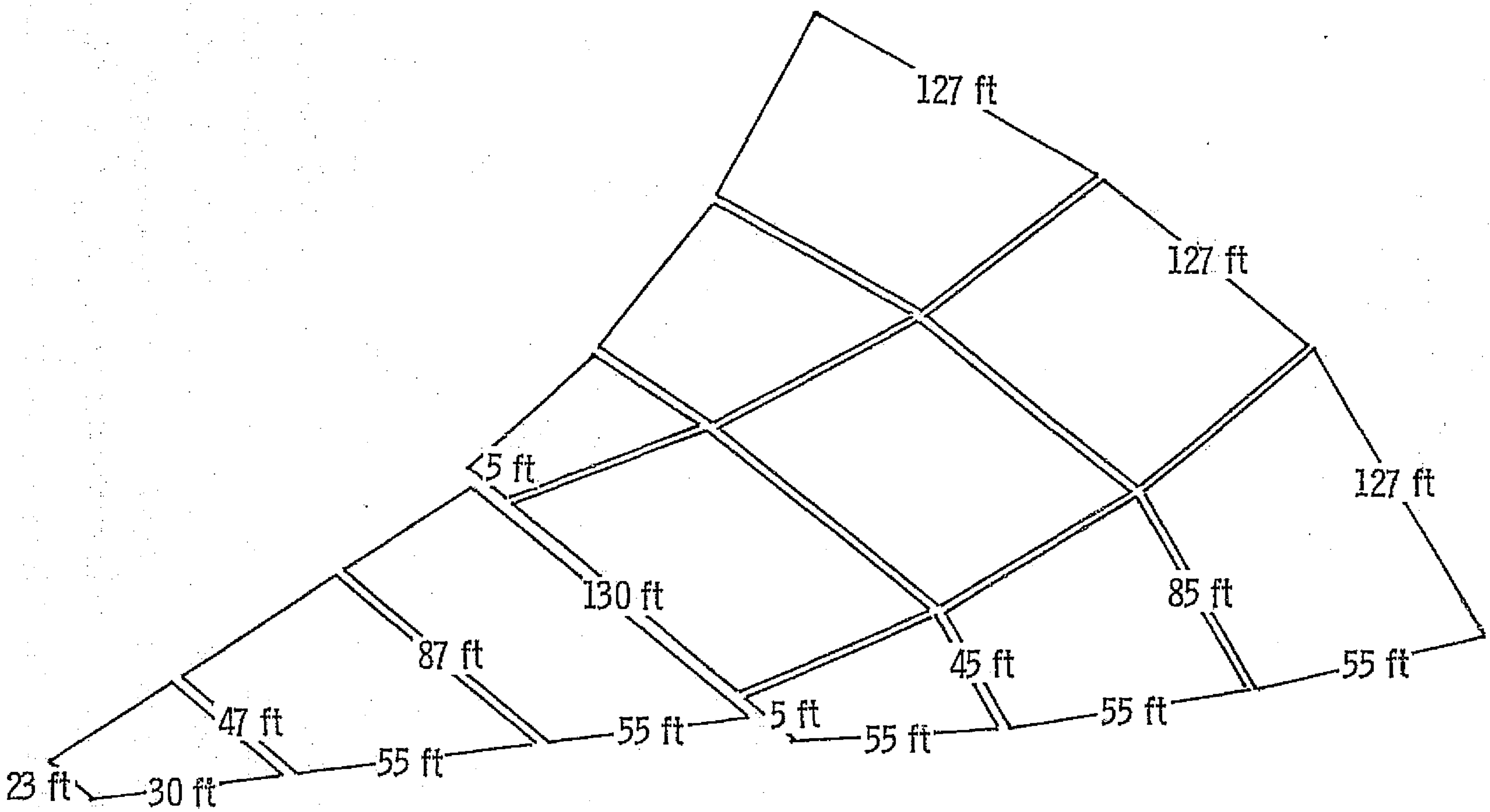

Figure IIIB-4 Radio Astronomy Telescope Mesh Assembly

the two rib sections. Figure IIIB-5 shows how these three beams fit together and roll within the next beam. Note the stowage of nets B and C. Beams 4, 5, and 6 are folded together and all three folded back over telescoped beams 1, 2, and 3. Some of the net will deploy automatically as the beams fold out. Net which does not deploy automatically must be physically pulled out of the respective beam (some roll out, some fold out) and attached to the opposite beam. Figure IIIB-4 shows the net pattern over the ribs. Even though they are shown as separate blocks, they are actually electrically connected together through their aluminum beams. Note the circumferential wires located at the mid-diameter and outer diameter of the ribs (Figure IIIB-3). These cables will provide stiffness if forces are bending the ribs aft, in which case the cables become tensioned.

C. PROCEDURES AND TECHNIQUES (TASK 5), AND TRADEOFFS (TASKS 3 AND 4)

Surely all assembly in LEO or full assembly in HEO (manned and unmanned) are logical and straightforward thoughts, while partial assembly in LEO and final assembly in HEO is an obvious combination of these. In the first part of the study, we considered full assembly in LEO with Tug boost to HEO (Approach 1), full assembly in HEO without man (Approach 2), and full assembly in HEO with man (Approach 3). Partial assembly in LEO and partial assembly in HEO were not analyzed since this introduces many possible combinations and presents no problems not encountered in the three approaches we analyzed.

The following is a detailed description of each of the approaches with the corresponding thought patterns which predicated some design changes to match the configuration with the assembly approach logistical problems. A comparison of the approaches will be presented in part 4 of this section.

1. Assembly Approach 1

a. Description - For this approach, all Radio Astronomy Telescope assembly and checkout is completed in the 160 n mi

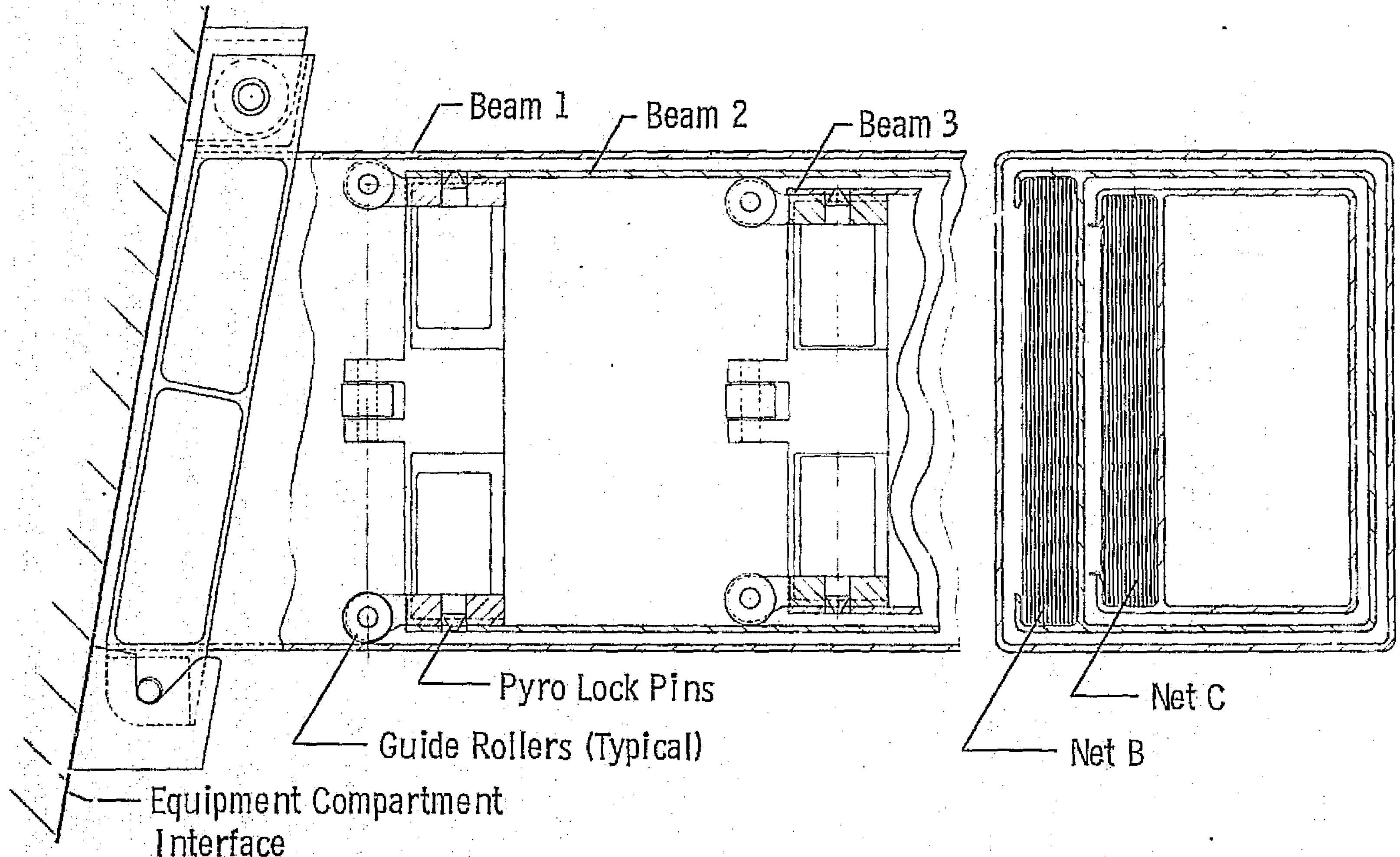

Figure IIIB-5 Radio Astronomy Telescope Beam Section

Shuttle orbit (see Figure IIIC-1). This allows full use of the Shuttle capabilities, such as man's presence for EVA and the use of the RMS for assembly. Shuttle flight 1 takes the mast assembly to LEO, and deploys and checks it out. The deployed mast requires temporary attitude control thrusters and propellant for stabilization between Shuttle flights. Shuttle flight 2 contains the beam/mesh assemblies and a docking module. After docking, the beams are emplaced with the RMS. After checkout, the telescope is released and the beams totally deployed. Two EVA astronauts, using MMUs for translation, will deploy the tension cables and mesh panels. After checkout, the beams are folded at the midpoints in preparation for Tug boost. Shuttle flight 3 contains a Tug. The Tug is docked with the telescope and boosted to HEO where the beams are hinged out, the total telescope is verified operational, and pointed and released from Tug. The Tug returns to Shuttle for return to earth.

b. Discussion - Approach 1 is the most straighforward approach. It features total satellite assembly in LEO which has many inherent advantages:

1) Assembly is in the vicinity of Shuttle. This allows full use of all Shuttle capabilities as described previously and avoids the use of either a manned Tug or a sophisticated, remotely controlled free flyer.

2) This approach affords the unique advantage of complete checkout in LEO, thus avoiding the launch of a Tug only to find the space system has a malfunction when in HEO.

Approach 1 also has a number of disadvantages due primarily to the need for the Tug to boost the assembled satellite to HEO. The basic Tug capability is discussed in Appendix A where it is seen that the baseline thrust is 15,000 lbs which imposes a significant load to the telescope structure. An analysis was done to determine if an excessive deflection (one that could be injurious to the structure) was indicated. Analysis showed that the fully deployed satellite could not withstand the 15,000 lb force, which yielded .312 g's at thrust tail-off.

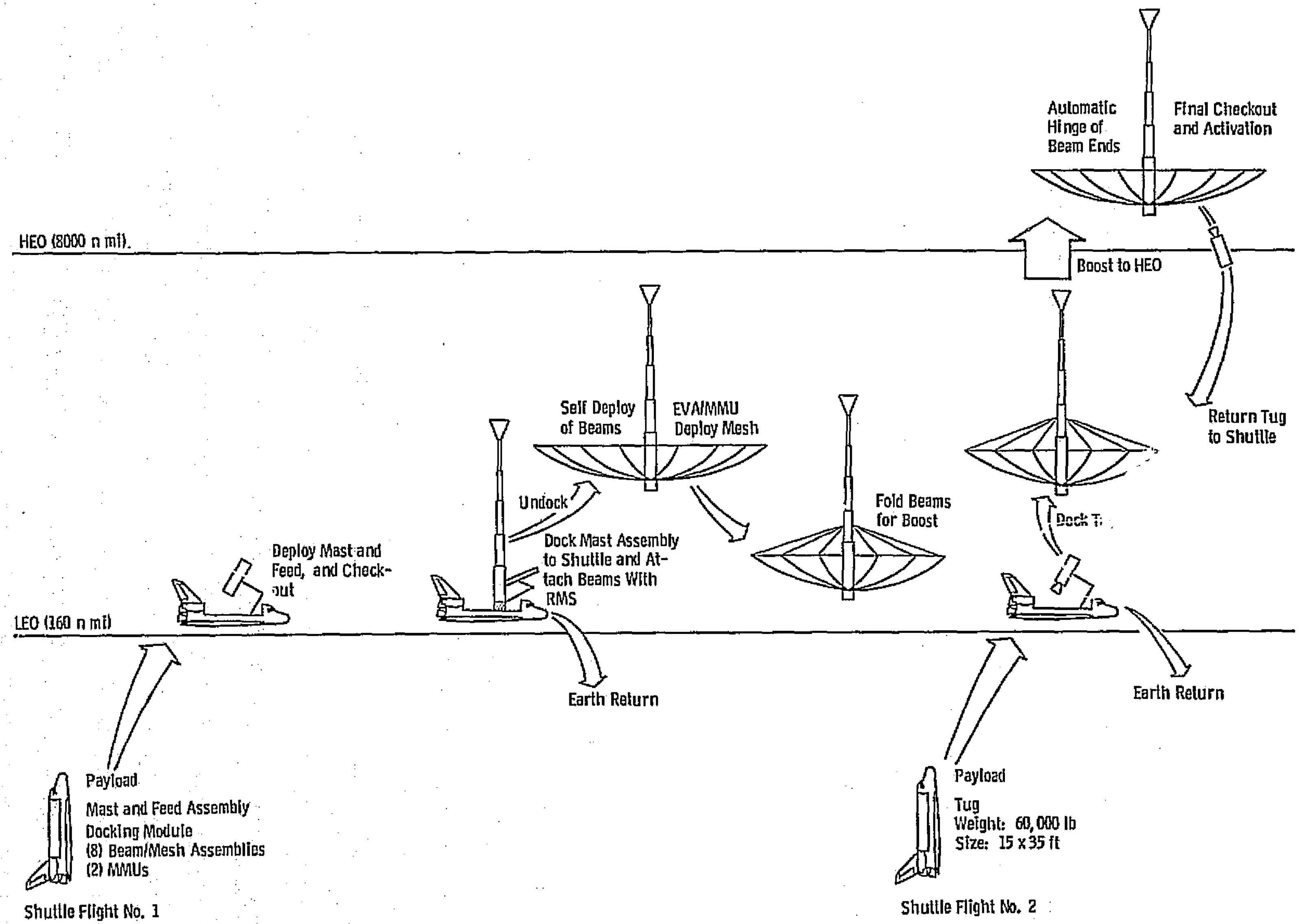

Figure IIIC-1 Radio Astronomy Telescope Assembly, Approach 1 (LEO Assembly)

Two or three solutions presented themselves at this time: 1) refolding the deployed satellite into a more compact configuration and using cables to increase stiffness, or 2) lowering the thrust from 15,000 lbs to $\sim$4,000 lbs which results in a substantially lower maximum acceleration of .082 g's. Both can theoretically lift the 25,000 to 30,000 lb payload into the proper 8,000 n mi orbit, although efficiency is lost as the thrust levels get lower despite the impulse being virtually the same. It was found that a refolding of the structure and use of cables would satisfy the deflection requirement.

It is possible with beam redesign to increase stiffness and reduce deflection, but the volumetric constraints exerted on the design by Shuttle will result in a much higher weight and one then finds the satellite weight exceeding the Tug maximum payload. One possibility here is to use two Tugs in tandem for the boost to HEO. However, each Tug brought to LEO requires one extra Shuttle and this represents a great delta increase in cost.

Deflection, however, is not the only problem. There may be severe dynamical vibrations induced into the satellite due to the Tug boost which may have an even more deleterious effect than deflections. The analysis of these phenomena require a major effort and is beyond the scope of this contract. In addition, this design at present is just an example.

As the telescope is being boosted into HEO, the Tug's ACS will be used to control the Tug/telescope combination in attitude. This can become a problem since the Tug's gimballed engine has only a 3 deg rotational capability. The c.g. of the Tug/telescope combination must be maintained within a five foot cone around the c.g. location (see Figure IIIC-2 for a pictorial description of the refolded satellite being boosted by the Tug). Analysis shows that this can easily be accomplished based on the configuration at this time. (Figure IIIC-7 is one example of several folding approaches being considered.)

It can be seen from this discussion that Approach 1 can start to become unwieldy as the satellite grows in size and weight and a general approach toward assembly in space dictates the necessity for considering approaches 2 and 3.

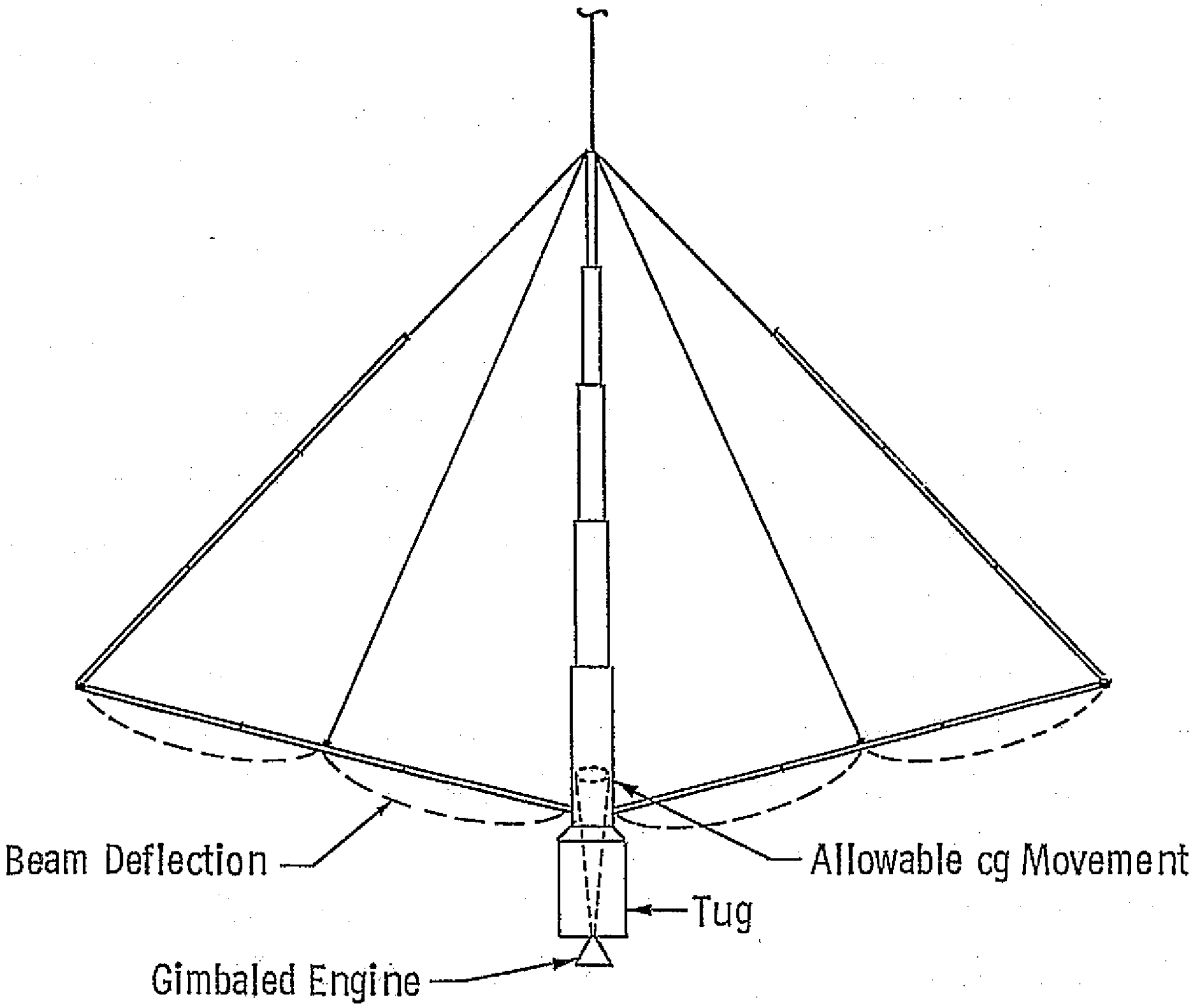

Figure IIIC-2 Tug Boosting Assembled Satellite to HEO: Potential Problems

c. Scenario - The following is a detailed scenario for approach 1. It is shown in chronological order starting with the launch of each of the three Shuttle flights (A, B, and C), its cargo, and subsequent deployment procedure up to the checkout of the assembly in HEO.

1. Shuttle Flight A

a. Cargo - 15 x 60 ft center can, weight approximately 10,000 lbs which contains:

- Beam attach mechanism;
- 15 x 60 structural can;
- Docking mechanism;
- Tower and feed assembly and deployment mechanism (stowed);
- Instrumentation ACS and temporary propulsion system;
- EOTS and two MMU's.

b. Deployment Sequence -

A-1. Achieve LEO.
A-2. Checkout RAT subsystems in cargo bay via umbilical (from orbiter payload specialist station).
A-3. Deploy out of cargo bay with RMS.
A-4. Checkout temporary ACS.
A-5. Position radio astronomy telescope mast/can for deployment.
A-6. Deploy center mast and feed, using orbiter electrical power.
A-7. Checkout and verify radio astronomy telescope subsystems and solar cells via S-band antenna and feed network.
A-8. Disconnect power umbilical.
A-9. Position radio astronomy telescope mast for release.
A-10. Release RMS/radio astronomy telescope.
A-11. Back off orbiter.
A-12. Inspect radio astronomy telescope mast (visually) to insure structural integrity and docking mechanism.
A-13. Verify subsystem operations and ACS attitude hold capability.

2. Shuttle Flight B

a. Cargo -

- Shuttle docking module, weight (TBD);
- Eight beam/reflector mesh assemblies, 1500 lbs ea - 12,000 lbs total;
- Possible extra OMS kits, weight (TBD) and size (TBD);
- EOTS;
- MMU;
- Propulsion packages (four), and two star trackers.

b. Deployment -

B-1. Achieve LEO.
B-2. Rendezvous with radio astronomy telescope mast assembly.
B-3. Place radio astronomy telescope mast in fine attitude hold.
B-4. Align Shuttle for docking.
B-5. Dock Shuttle with radio astronomy telescope, verify, and deactivate [consider subsystem module exchange at this time (checkout)] radio astronomy telescope, ACS.
B-5a. Connect and activate instrumentation and power umbilical.
B-6. Activate RMSs.
B-7. Detach beam holddown(s), repeat for each.
B-8. Attach RMS to beam assembly at predetermined point.
B-9. Translate beam (#1) and align with attach point on radio astronomy telescope center structure.
B-10. Attach beam end and verify.
B-11. Activate pyro-bolt attach mechanism.
B-12. Attach beam (#1) electrical umbilical, verify (could be internal).
B-13. Repeat B-7 through B-12 for beams #2, #3 and #4.
B-14. Attach structural cable (5), (2) from mast to beam ends (55 ft). (Repeat for each beam.)
B-15. Rotate beam ends (Y segments) up to 90 deg for beams 1 through 4.
B-16. Using RMS, locate, attach to, and extend to next beam reflector mesh (on first 55 ft segments only).
B-17. Repeat B-16 for beams #2, #3, and #4.
B-18. Using RMS, detach holddown mechanism on beam structural cross members and rotate member to align with adjoining beam.
B-19. Attach latching mechanism on cross member end, verify and activate pyro-bolt.
B-20. Repeat B-14 and -15 for beams #2, #3, and #4.
B-21. Inspect and verify all assembly.
B-22. Detach main instrumentation and power umbilical.
B-23. Activate and verify radio astronomy telescope ACS (propellant may require resupply).
B-24. Undock radio astronomy telescope from Orbiter.
B-25. Rotate Orbiter 180 deg (check CG offset problem). Could hold with RMS.

B-26. Align Shuttle for docking.

B-27. Dock Shuttle with radio astronomy telescope, verify and deactivate radio astronomy telescope ACS.

B-28. Connect and activate main instrumentation and power umbilical.

B-29. Repeat B-6 through B-19 for beams #5, #6, #7, and #8.

B-30. Detach main instrumentation and power umbilical.

B-31. Activate and verify radio astronomy telescope ACS.

B-32. Undock Shuttle from radio astronomy telescope.

B-33. Translate Shuttle to TBD m from radio astronomy telescope and stabilize.

B-34. Activate each beam (8) to full extension (one at a time).

B-35. Activate pyro-bolts at beam joints, all except at "Y" hinge point.

B-36. Inspect and verify beam assemblies.

B-37. Activate and checkout MMU/EVA astronaut(s).

B-38. Translate MMU to radio astronomy telescope beam assemblies.

B-39. Position MMU to predetermined point on beam segments.

B-40. Astronaut grasp beam cable end(s).

B-41. Translate MMU to cable receptacle on opposite beam.

B-42. Attach cable end, tension, and verify.

B-43. Repeat B-36 through B-40 for approximately 32 structural/segments cables between the eight beams (may require several resupplies for MMU propellant and batteries).

B-44. Inspect and verify all attachments.

B-45. Translate MMU to Shuttle cargo bay, dock and resupply.

B-46. Activate two MMU/EVA astronauts and stem deployment mechanism.

B-47. Translate MMUs to beam segments requiring mesh deployment.

B-48. Position astronaut/MMUs on each side of beam segments (MMU not docked with beam).

B-49. Astronaut (1) positions stem mechanism on beam receptacle.

B-50. Astronaut (1) activates stem and translates attachment mechanism to opposite beam.

B-51. Astronaut (2) monitors alignment and attachment and aids (as required).

B-52. Astronaut (1) retracts stem with reflector mesh segment and attaches to beam.

B-53. Astronaut (1) inspects and verifies mesh attachments and electrical connections.

B-54. Astronaut/MMUs (1 and 2) translate to next beam segment.

B-55. Repeat B-46 to B-51 for 248 reflector panels (may require many MMU resupplies).

B-56. Translate MMUs to cargo bay, dock and resupply.

B-57. Translate propulsion packages (4) to radio astronomy telescope from Shuttle cargo bay.

B-58. Dock or restrain MMU at "Y" intersection at middle of beam.

B-59. Attach propulsion package (approximately 2 ft^2 and (TBD) lbs) both structurally and electrically.

B-60. Checkout and verify operation of thrusters.

B-61. Repeat B-55 through B-58 for three other thruster packages.

B-62. Translate to Shuttle and pick up two star tracker packages.

B-63. Translate to radio astronomy telescope and dock MMU at "Y" intersection at middle of beam opposite beam used for thruster packages.

B-64. Attach star tracker package (1), checkout and verify (weight: (TBD) lbs; approximately 1 x 1 x 2 ft).

B-65. Translate to beam 180 deg from package (1) and dock.

B-66. Attach star tracker package (2), checkout and verify.

B-67. Translate MMUs to cargo bay, dock, stow and resupply.

B-68. Inspect and check out complete radio assembly telescope assembly.

B-69. Via remote control, activate internal beam joint drivers to rotate beam "Y" joints up 90 deg $\pm$ and verify operation.

B-70. Activate MMU and translate to upper radio astronomy telescope assembly.

B-71. Attach eight beam ends using temporary attachment device.

B-72. Return MMU to Shuttle, dock, and stow.

B-73. Place radio astronomy telescope in standby mode and verify.

3. Shuttle Flight C

a. Cargo –

- Tug, 15 x 35 ft, 60,000 lbs;
- One (1) EOTS;
- Two (2) MMUs.

b. Deployment Sequence -

C-1. Achieve LEO.
C-2. Rendezvous with radio astronomy telescope and standoff (TBD) m.
C-3. Checkout Tug in cargo bay.
C-4. Checkout radio astronomy telescope and place in fine attitude hold.
C-5. Activate RMS and deploy Tug.
C-6. Translate Tug to radio astronomy telescope docking receptacle and align.
C-7. Dock Tug and radio astronomy telescope and verify.
C-8. Perform complete systems radio astronomy telescope/Tug.
C-9. Align radio astronomy telescope/Tug for boost, stabilize and verify.
C-10. Back off Shuttle to (TBD) n mi from radio astronomy telescope/Tug.
C-11. Boost Tug/radio astronomy telescope to HEO.
C-12. Verify orbit and make corrections if necessary.
C-13. Position radio astronomy telescope and stabilize.
C-14. Separate Tug from radio astronomy telescope and translate (TBD) n mi from radio astronomy telescope.
C-15. Activate and checkout radio astronomy telescope subsystems.
C-16. Translate Tug to LEO, rendezvous with Shuttle, RMS capture and stow in cargo bay.
C-17. Detach temporary beam holddowns (from Shuttle or ground).
C-18. Activate all beam joint rotation devices simultaneously.
C-19. Verify beam joint locations.
C-20. Fire pyro-bolts to lock beam joints.
C-21. Make final checkout and verification of all subsystems and pointing and tracking systems.

2. Assembly Approach 2

a. Description - For this approach, all radio Astronomy Telescope assembly is done in HEO without man present, except remotely from LEO or the ground (see Figure IIIC-3). Shuttle flight 1 contains the mast/feed assembly. It is met in LEO by Shuttle flight 2, which contains a Tug. These are mated so the Tug attitude control system can stabilize the entire package.

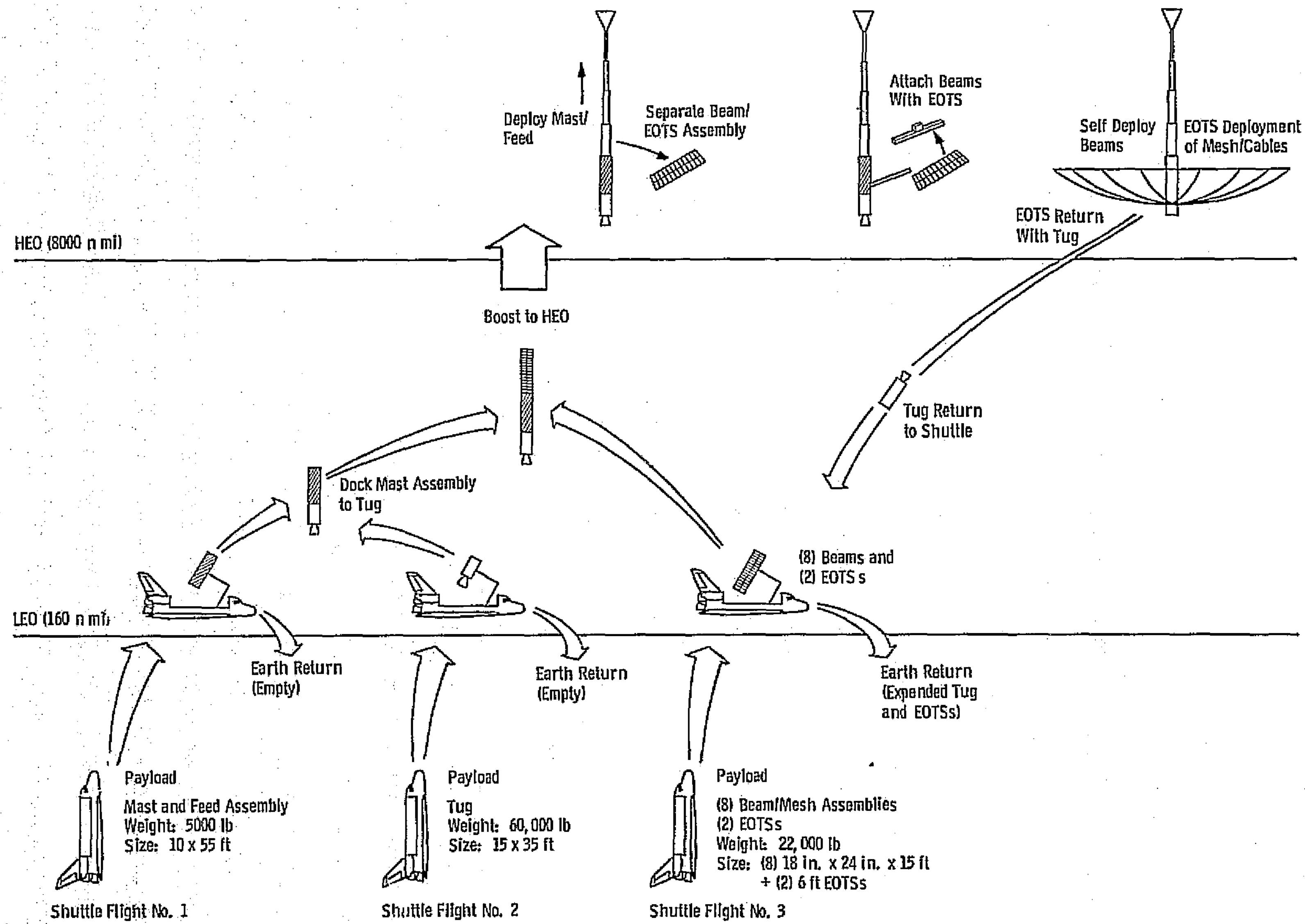

Figure IIIC-3 Radio Astronomy Telescope Assembly Approach 2 (HEO Assembly "Unmanned")

Shuttle flight 3 contains the beam/mesh package and two EOTSs. This assembly is attached to the orbiting Tug/mast and boosted to HEO. The beam package is separated from the mast and attitude control is provided by the EOTSs. The mast and feed are deployed. The EOTSs translate and attach the separate beam assemblies. The beams self-deploy. The EOTSs deploy the cables and mesh. After operational verification, the EOTS docks with the Tug and returns to LEO and Shuttle returns to earth.

b. Discussion - Approach 2 has the inherent advantage over Approach 1 of no potential deflections, vibrations or maximum payload problems. This is due to the fact that the Tug will boost up the individual parts all in tandem. To do the assembly in this way implies a much greater number of docking mechanisms during boost to HEO. Another problem is that the assembly must be done without Shuttle nearby to serve as a platform, or lend help with an RMS. It has the disadvantage of no LEO checkout in that the system will not give any indication of its operational state until assembled in HEO. The primary problem encountered, however, is the reliance on the free flyer and Tug to perform all the assembly tasks that were done by man in Approach 1.

3. Assembly Approach 3

a. Description - For this approach, all Radio Astronomy Telescope assembly is done in HEO with man present (see Figure IIIC-4). Shuttle flights 1, 2, and 3 take the Radio Astronomy Telescope/Tug to LEO. Tug boost and beam assemblies are all conducted in the same manner as in approach 2. Because the deployment of the cables and mesh may be very difficult to accomplish with an EOTS, approach 3 uses EVA astronauts in MMUs in HEO. This requires a manned Tug and a dedicated Shuttle (4th) flight.

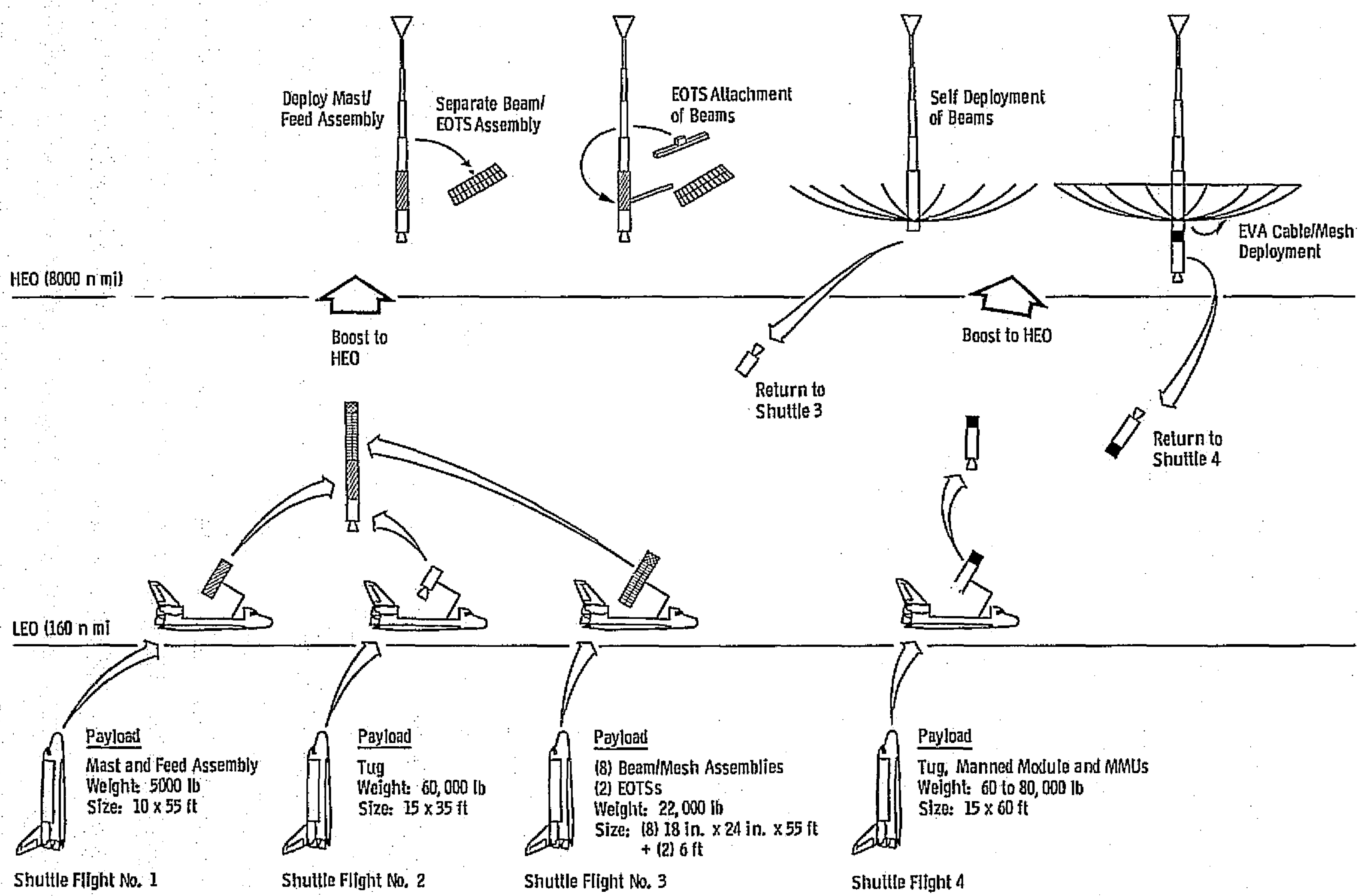

Figure IIIC-4 Radio Astronomy Telescope Assembly Approach 3 (HEO Assembly "Manned")

b. Discussion - This approach is very similar to approach 2 but will require a manned EVA in HEO. This demands a manned Tug (which at this time is still only a concept) and, in addition, there is a greater risk for EVA in HEO, beyond that associated with EVA in LEO. The resupply and refurbishment problem from the Tug is a major one and has yet to be addressed. Another disadvantage is the need for one extra manned Tug and one extra Shuttle to boost it to LEO. This in itself represents a significant difference in cost.

4. Assembly Approach Comparisons

The following is a comparison of the three approaches featuring a cost and non-cost analysis.

The cost table (see Table IIIC-1) demonstrates that for a constant satellite development cost which is almost always the prime cost component, the delta costs are generally seen to be in the Shuttle flight areas as well as in the special mechanisms or specialized training areas. Support equipment of the standard variety such as EOTS, MMU, and MSM are expensive and sophisticated, but it is felt that these are going to be employed by many users thus spreading the cost to one satellite down to a fairly low figure.

Table IIIC-1 Cost Comparisons for Assembly Approaches (dollars in millions)

	1 Assembly in in LEO		2 Assembly in HEO, Unmanned		3 Assembly in HEO, Manned	
Shuttle Flights	20.0		20.0		30.0	
Tug Flights	1.0		1.0		2.0	
Satellite Development	56.1		56.1		56.1	
Training	1.5		0.8		2.0	
Shuttle and Tug Adapter Mechanism	2.2		3.2		3.2	
Subtotal	80.8		81.1		93.3	
Support Equipment	A	B	A	B	A	B
• MMU	1.25	15.0	–	–	1.25	15.0
• EOTS	–	–	.32	48.4	.32	48.4
• MSM	–	–	–	–	.9	135.0
TOTALS	82.05	95.8	81.42	129.5	95.77	291.7

A = Assumes this equipment is used for maintenance of all satellites;
B = Assembly of Radio Telescope must absorb entire cost of equipment.

Both the spread costs and the costs assuming full accountability to the space system are presented.

Shuttle flight costs of $10M were used while Tugs were assumed to be $1M for flight costs. The satellite development price of $56.1M was generated from a computer program administered by Bill Haldeman at JSC. It considered a detailed description of the gear, the fact that this was a new design and not a modification of an existing approach, the kind of materials being used, a schedule of design, development test, qualification and fabrication of the first units and the level of documentation that will be used during the Phase C program.

The training costs were generated by MMC, specifically from our M-509 area where the people are quite familiar with the necessary operations and the level of effort needed to support this task. Shuttle and Tug adapter mechanisms costs were scaled from the McDonnell-Douglas PUT study in which these adapters were initially costed. Approaches 2 and 3 will require a number of adapters since the parts will be unassembled during Tug boost to HEO and, of course, show a higher price. The subtotals are very similar, with the basic differences being in the number of Shuttles used.

It is in the area of support equipment that the costs become large and somewhat subjective to deal with. MMUs used as an EVA aid are presently being considered as support equipment for twelve Shuttle payloads. The $15M cost for the D, DT&E and the spread cost over 12 uses is shown in the table. The EOTS and MSM costs which are quite large are shown spread over 150 users whose payload and servicing operations can be handled by these support equipment once developed.

The non-cost comparison of the three assembly approaches is presented in Table IIIC-2. These seven items were chosen as the most representative factors to be analyzed due to their contribution to overall mission success. They are quite general but form a good basis to compare assembly approaches.

A subjective weighting scheme is shown which compares each of the seven items with respect to each other with the higher values corresponding to the more significant items. A unit rating is then assigned which numerically compares the three

Table IIIC-2 Non-Cost Comparisons of Assembly Approaches for the Radio Astronomy Telescope

		Approach 1 Assembly in LEO		Approach 2 Assembly in HEO Unmanned		Approach 3 Assembly in HEO Manned	
	Weight	Unit Rating	Total Rating	Unit Rating	Total Rating	Unit Rating	Total Rating
A. Man Safety	25	1	25	4	100	8	200
B. Assembly Reliability	20	2	40	5	100	4	80
C. Support Equipment Complexity and Development Program	15	2	30	9	135	5	75
D. Equipment Safety	15	2	30	5	75	5	75
E. Potential Problems in Transit	13	7	91	2	26	2	26
F. Mission Complexity	7	3	21	9	63	6	42
G. Mechanical Complexity	5	3	15	8	40	5	25
TOTALS	100		252		539		523

approaches for each item. A "1" is considered the best rating while a "10" is considered the worst.

It is not surprising that approach 1 appears to be best both from a cost and non-cost aspect. The primary problem with approach 1 will be its difficulty in being boosted to HEO after assembly. It is felt that by refolding and using cables for extra support, the deflections/vibrations problem can be minimized and the approach implemented.

5. Transportation

One of the major difficulties encountered with assembly of the Radio Astronomy Telescope centers around the altitude in which it is to be performed. High earth orbit either manned or unmanned is inherently a problem in that new and sophisticated

hardware must be developed to support the many assembly requirements. Low earth orbit (Shuttle orbit) assembly offers many advantages from the standpoint of having Shuttle nearby and the relative ease of incorporating manned functions. One of its negative factors is the necessity for Tug boost after the telescope is assembled which could cause structural problems due to its large size and high thrust level. The use of the Solar Electric Propulsion Stage (SEPS), a very low thrust booster normally intended for high altitude utilization, was considered, recognizing its time limitations, as an alternative to Tug. SEPS was shown to have sufficient thrust capability (.206 lb) to boost a satellite to an 8,000 to 10,000 n mi orbit if a number of aspects could be satisfied:

- Disturbances
  - Aerodynamic
  - Solar Pressure
  - Gravity Gradient
- Obscuration from the sun, by both the earth and by the assembly itself
- Van Allen belt radiation degradation of SEPS solar cells

Since it will be shown that the radiation flux seriously degrades the SEPS solar panels, a discussion of the geometry of the Van Allen belt is presented. Early indicators identified a problem with the radiation flux and we strove to find an orbit especially in the low altitude region that would avoid the problem areas. Figure IIIC-5 demonstrates that the predominance of the flux is at higher altitudes (1,000 to 8,000 miles) with peaks at 2,000 to 3,000 miles. But the South Atlantic anomaly represented by 10^6 flux contours at varying altitudes shown in Figure IIIC-6 grows very quickly in size once into the 500 to 1,000 mile range, demonstrating a situation impossible to avoid by careful choice of orbital parameters.

This discussion highlights the major reason why SEPS cannot be used to boost the assembled telescope from low earth

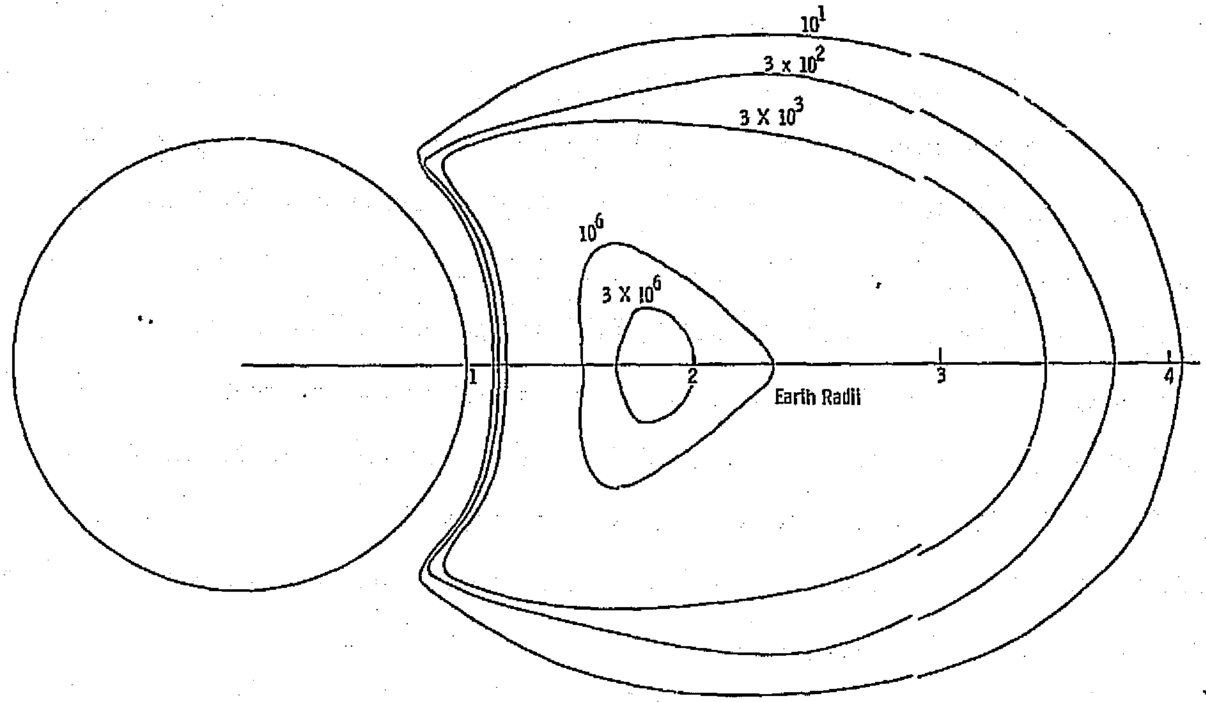

Figure IIIC-5 Uniform Radiation Flux for Simplified Dipole Model

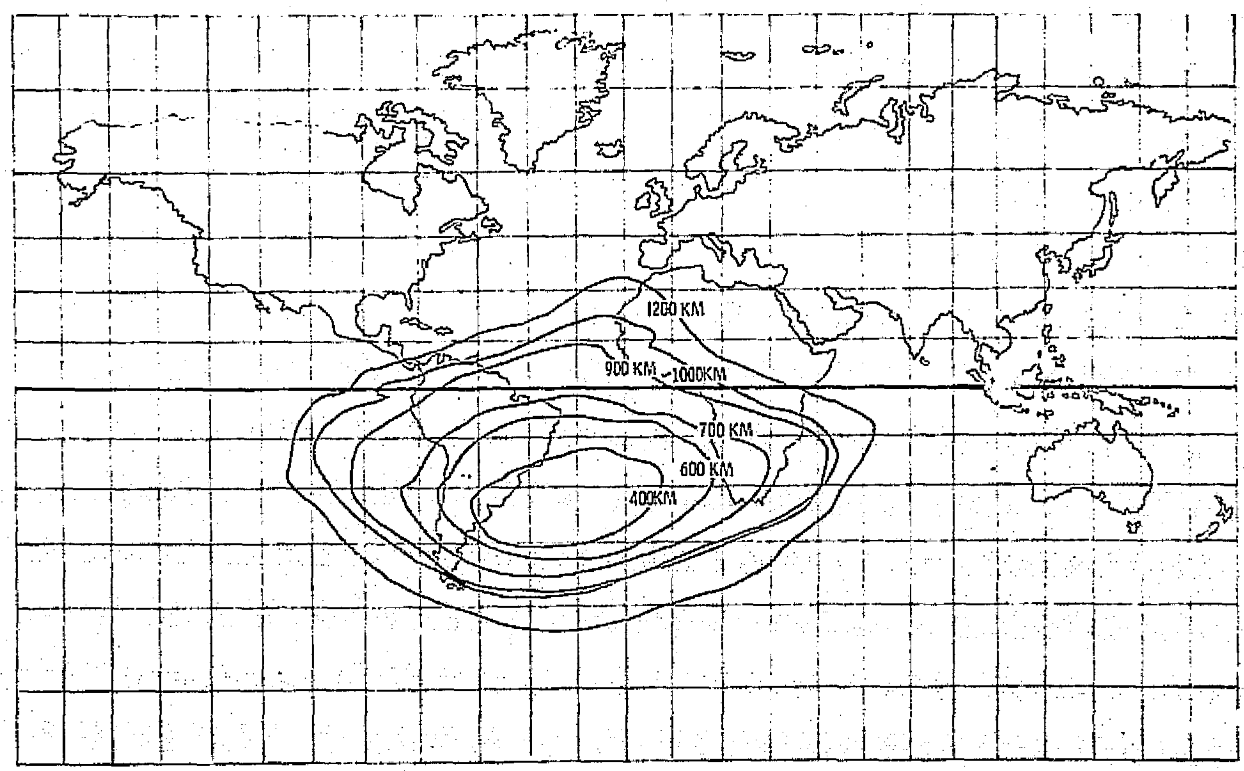

Figure IIIC-6 South Atlantic Anomaly Radiation Flux vs Altitude

orbit to any applicable higher altitude. The MAPSEPS* computer runs demonstrated that under no circumstances could SEPS be considered as a boosting vehicle in these low altitudes. The conclusion is a function of the NASA-LRC developed radiation flux model which is used in the MAPSEPS program.

There was considerable thought given to possible alternatives which would alleviate this problem area:

1) Better shielding of the solar panels to avoid power degradation. Higher shielding of the solar panels at this time was shown to be quite heavy and not effective enough to minimize solar panel damage to the point of utility.

2) Rolling the panels in to avoid troublesome areas such as the anomaly at low altitudes loses its effectivity due to the ubiquity of the radiation particles at higher altitudes. It is also a very difficult task to effect mechanically with any reasonable reliability.

3) Nuclear electric propulsion, while outside the scope of this program, appears to merit some future consideration in its ability to provide a similar low thrust over long periods of time very efficiently. It suffers from none of the major problems encountered by SEPS such as radiation damage and shadowing, but may present other problem areas such as contamination.

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* MAPSEPS is a program developed by MMC for NASA-MSFC which, among other things, can ascertain the amount of degradation and shadowing experienced by a vehicle being boosted by solar propulsion. It keeps track of the orbital parameters and demonstrates the performance loss due to these limiting factors. It has been used in this program to show SEPS inability to help in low orbit.

IV. MAINTENANCE

A. INTRODUCTION

The objectives of the maintenance portion of this study were to investigate, further develop, and assess technical and operational concepts for the manned and automated maintenance of seven satellites. Two of these satellites were to be the subjects selected for the assembly portion of this study (radio astronomy telescope (RAT) and the microwave power transmission system (MPTS) antenna from the solar power station). The MPTS was subsequently dropped from detailed maintenance analyses because of lack of sufficient data on the antenna operational subsystems. Another satellite to be investigated was the Earth Observations Geosynchronous Platform discussed in the NASA funded study, Geosynchronous Platform Definition Study (NAS9-12909). The remaining four satellites were to be selected from the 17 geosynchronous satellites defined in the 1973 Shuttle Traffic Model, NASA TM X-64751, Revision 2.

Related studies and other supporting documents were to be considered and used in this study to avoid duplication of effort. In reviewing other studies, it became apparent that there are a multitude of potential satellite configurations in regards to the methods for locating replaceable units. It was therefore suggested that a unique and desirable output of this maintenance study should be the investigation of maintenance requirements from the standpoint of different satellite serviceable configurations. Another desired study result would be design criteria for a universal servicer applicable to any of the satellite configurations studied and capable of maintenance/replacement of equipment outside of a standard fixed location (such as solar arrays).

An additional task (Task 6) was to investigate the feasibility of an on-orbit automated maintenance vehicle that can remain in geosynchronous orbit for an extended time and perform maintenance operations. The results of this task were compared to the other maintenance approaches analyzed.

The study plan was revised to incorporate the previously discussed items. In general, the following steps were accomplished in the maintenance portion of this study.

1) Select geosynchronous satellites to be studied.

2) Compile baseline data on each satellite to be studied.

3) Reconfigure each satellite to a serviceable version (where applicable). Each satellite was configured in a different way, corresponding to various proposed configurations from previous studies.

4) Prepare detailed procedures (scenarios) and analyze techniques and requirements for various maintenance approaches.

5) Develop tradeoff data and compile advantages and disadvantages of each approach.

6) Develop servicer requirements for each of the satellites.

7) Develop the conceptual design requirements for a general purpose universal servicer.

8) Determine need for simulations of various maintenance tasks. (The decision was subsequently made that no simulations of maintenance tasks was justified at this time.)

9) Conduct analysis of on-orbit geosynchronous maintenance vehicle.

For purposes of this study, the ground rules presented in Table IVA-1 are assumed.

B. REQUIREMENTS AND SATELLITE SELECTION (TASK 1)

Information was reviewed on the 17 identified geosynchronous satellites (Table IVB-1) from the Space Shuttle Payloads Description (SSPD) documents[1]. Data were compiled on the satellite schedules, quantities and sizes, mission equipment, and supporting subsystems. Several of the satellites were eliminated from further consideration because they were low cost expendable (LCE) items, they were similar in configuration to other satellites, or insufficient data were available. From the remaining nine satellites, presented in Table IVB-2, a judgmental selection was made of the four to be analyzed in the maintenance study.

With concurrence of the NASA Contracting Officer's Representative, the four selected geosynchronous satellites were:

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[1] Summarized NASA Payloads Descriptions, Automated Payloads, Level A Data, MSFC, July 1974 and Payload Descriptions, Automated Payloads, Level B Data, MSFC, July 1974.

Table IVA-1 Ground Rules for Maintenance Studies

1. Maintenance is defined as restoration of functional capabilities or updating of system capabilities. Verification checkout is included. Servicing is one form of maintenance.
2. The maintenance need (failure, equipment updating, consumables replenishment, etc.) is identifiable at the ground and a maintenance mission may be effected.
3. The following support equipment is available:
   a) Shuttle Remote Manipulator System (RMS)
   b) RMS with manned platform (arms controllable from platform)
   c) Extravehicular Activity (EVA)
   d) Manned Maneuvering Unit (MMU)
   e) Earth Orbital Teleoperator System (EOTS)
   f) Servicer (attached to a mother vehicle)
   g) TUG (unmanned)
   h) Interim Upper Stage (IUS) (one way TUG)
   i) Manned TUG
   j) Solar Electric Propulsion Stage (SEPS) (considered only in the SEPS part of the study)
4. The design of the satellite and servicing device is assumed to be adequate from the standpoints of docking, clearances, physical interfaces, simplicity, etc. to permit the successful completion of the maintenance tasks.
5. Any satellite subsystem for which failure may be reasonably anticipated, including appendages, should be assumed to be replaceable.
6. Ground refurbishment assumes facilities to restore the satellite to the original or better (updated) functional capability.
7. Maintenance in the payload (P/L) bay may include manned troubleshooting, subsystem checkout using orbiter equipment, and repairs with special equipment anticipated and launched on the orbiter.
8. No consideration is given to combining orbiter flights to achieve full payload capacity. Each maintenance mission will be evaluated only for its exclusive requirements.
9. The satellite to be serviced is assumed to be in orbit, having been placed there by the Shuttle/Tug.
10. Failures in support equipment are not considered.
11. Self-repair is another form of redundancy and is not considered.
12. The satellite must be docked to the servicing support equipment.
13. Tracking and Data Relay System (TDRS) satellites are available for relay of telecommunications.
14. Communications networks will allow real-time television of the workstation.
15. Communication with the satellite may be direct or through the IUS or TUG. The EOTS can only communicate with the Shuttle or TUG and not directly with the ground.
16. Electrical power to the satellite will be obtained from the TUG or orbiter during servicing. Satellite internal power will be off during servicing.
17. Electrical connections may be mated/demated during servicing. (Pin quantity is assumed minimized using data bus methods.)
18. Fluid connections may be mated/demated by quick disconnect methods if needed.

- Disaster Warning Satellite (DWS)
- U. S. DOMSAT C (Tracking and Data Relay Satellite - TDRS)
- Intelsat
- Synchronous Earth Observations Satellite (SEOS)

These four satellites were selected because they offered a good cross-section of characteristics of interest to the maintenance study. The characteristics considered were:

- Weight and size of the satellite
- Type and size of appendages
- Variety of mission equipment
- Variety of supporting subsystems
- Mission peculiarities
- Potential maintenance contamination problems

Table IVB-1 Geosynchronous Satellites

**1.	Advanced Radio Astronomy Explorer
2.	Advanced Synchronous Meteorological Satellite
*3.	Synchronous Earth Observatory Satellite
4.	Special Purpose Earth Observation Satellite
5.	Foreign Synchronous Meteorological Satellite
6.	Geosynchronous Operational Meteorological Satellite
7.	Geosynchronous Earth Resources Satellite
8.	Foreign Synchronous Earth Observatory
*9.	Intelsat
10.	U. S. Domsat A
11.	U. S. Domsat B
*12.	Disaster Warning Satellite
13.	Traffic Management Satellite
14.	Foreign Communications Satellite A
15.	Communications R&D Prototype Satellite
16.	Foreign Comsat B
*17.	U. S. Domsat C
*	Satellites recommended for further study.
**	Recommended alternate.

Table IVB-2 Satellites Retained After Initial Review

SATELLITE NAME	CODE NO.	WEIGHT CLASS (LBS)			TYPE ACS			DEPLOYED APPENDAGES				THERMAL CONTROL				SOLAR POWER		MISSION EQUIPMENT (QTY)	MISSION REQUIREMENTS	CONTAMINATION SUSCEPTIBILITY	DESIGNED FOR DOCKING	SERVICE PLANNED	RETRIEVAL PLANNED
		<1000	1000 TO 2000	>3000	N_2H_4	GN_2	CESIUM ION	SOLAR ARRAYS (SPAN)	DIPOLE ANTENNA	PARABOLIC ANTENNA	TELESCOPE COVERS	HEAT PIPES & RADIATOR	PASSIVE	LOUVERS	STRIP HEATER	ARRAY PANELS	BODY MOUNTED CELLS						
1. ADVANCED RADIO ASTRONOMY EXPLORER	AS-05-A (AST-1C)		X			X PLUS KICK MOTOR		X	X (FOUR 1475 FT)				X	X		X		5	2 IN ORBIT	ASPECT SENSOR	NO	NO	NO
3. SYNCHRONOUS EARTH OBSERVATORY	EO-09-A (EO-4)			X	X			X (21 FT)			X	X	X			X		4	2 IN ORBIT	OPTICS	YES	YES	YES
6. GEOSYNCHRONOUS OPERATIONAL METEOROLOGICAL	EO-58-A (EN/D-10)	X			X							X	X				X	8		OPTICS	YES	NO	NO
9. INTELSAT	CN-51-A (CN/D-1)			X	X		X	X (106 FT)					X			X		9	9 IN ORBIT	SENSORS	YES	NO	NO
10. U S. DOMSAT A	CN-52-A (CN/D-2A)	X			X								X				X	2		SENSORS	NO	NO	NO (LCB)
12. DISASTER WARNING SATELLITE	CN-54-A (CN/D-3)		X				X	X (130 FT)		X (19 FT)		X		X	X	X		2	2 IN ORBIT	SENSORS	YES	NO	NO
13. TRAFFIC MANAGEMENT SATELLITE	CN-55-A (CN/D-4)	X			X			X (42 FT)					X		X	X		3		SENSORS	YES	NO	NO (LCB)
16. FOREIGN COMMUNICATIONS SATELLITE	CN-56-A (CN/D-5)	X			X			X (30 FT)					X			X		7		SENSORS	YES	NO	NO
17. U. S. DOMSAT C	CN-58-A (CN/D-2C)		X		X			X (33 FT)		X (3 EA)			X	X	X	X		7	3 IN ORBIT (1 SPARE)	SENSORS	NO	TBD	TBD

The following discussions present brief overviews of the selected satellites. The reconfigured serviceable versions of these satellites will be discussed in the Conceptual Designs section (IVC) along with the EOGP and RAT.

Disaster Warning Satellite (DWS) - A view of this satellite is presented in Figure IVB-1. Data on the DWS is taken from Disaster Warning Satellite Study, TM X-68122, NASA-Lewis Research Center, March 1971.

There will be two of these satellites in geosynchronous orbit. They will relay disaster warning messages and bulletins from National Oceanic Atmospheric Agency (NOAA) ground sites to the U. S. public. The baseline satellite weighs about 1,284 lbs. The maximum operating power is 7,000 watts (when one satellite must also perform the transmission functions of the other satellite). Electrical power is derived from 900 ft^2 of solar cells located in two solar arrays. These arrays are rotated, by a single shaft, to always face the sun. Cross field amplifiers are used for the antenna transmission power. Heat is dissipated from these amplifiers, and other supporting equipment through the use of heat pipes, external radiators, and temperature controlled louvers. Eight cesium ion thrusters on the body and at the solar array tips are used to maintain orbit attitude and position.

U. S. DOMSAT C (Tracking and Data Relay Satellite - TDRS) - A view of the TDRS is presented in Figure IVB-2. This baseline configuration is the second alternate configuration developed in the Tracking and Data Relay Satellite System Configuration and Tradeoff Study (Part II), NASA CR-130218, Space Division, Rockwell International, April 1973. This configuration was chosen since it most closely corresponded to the Level A data presented in the 1974 SSPD.

There will be three of these satellites in orbit at the same time (two operating, one standby) to provide forward and return telecommunications links for low, medium, and high data-rate satellite users in earth orbit. The baseline satellite weighs about 671 lbs. Electrical power is derived from solar arrays rotated to always face the sun. Two hydrazine jet thruster quads are used to maintain orbit attitude and position. Temperature of equipment inside the body housing is primarily controlled by louvered radiators.

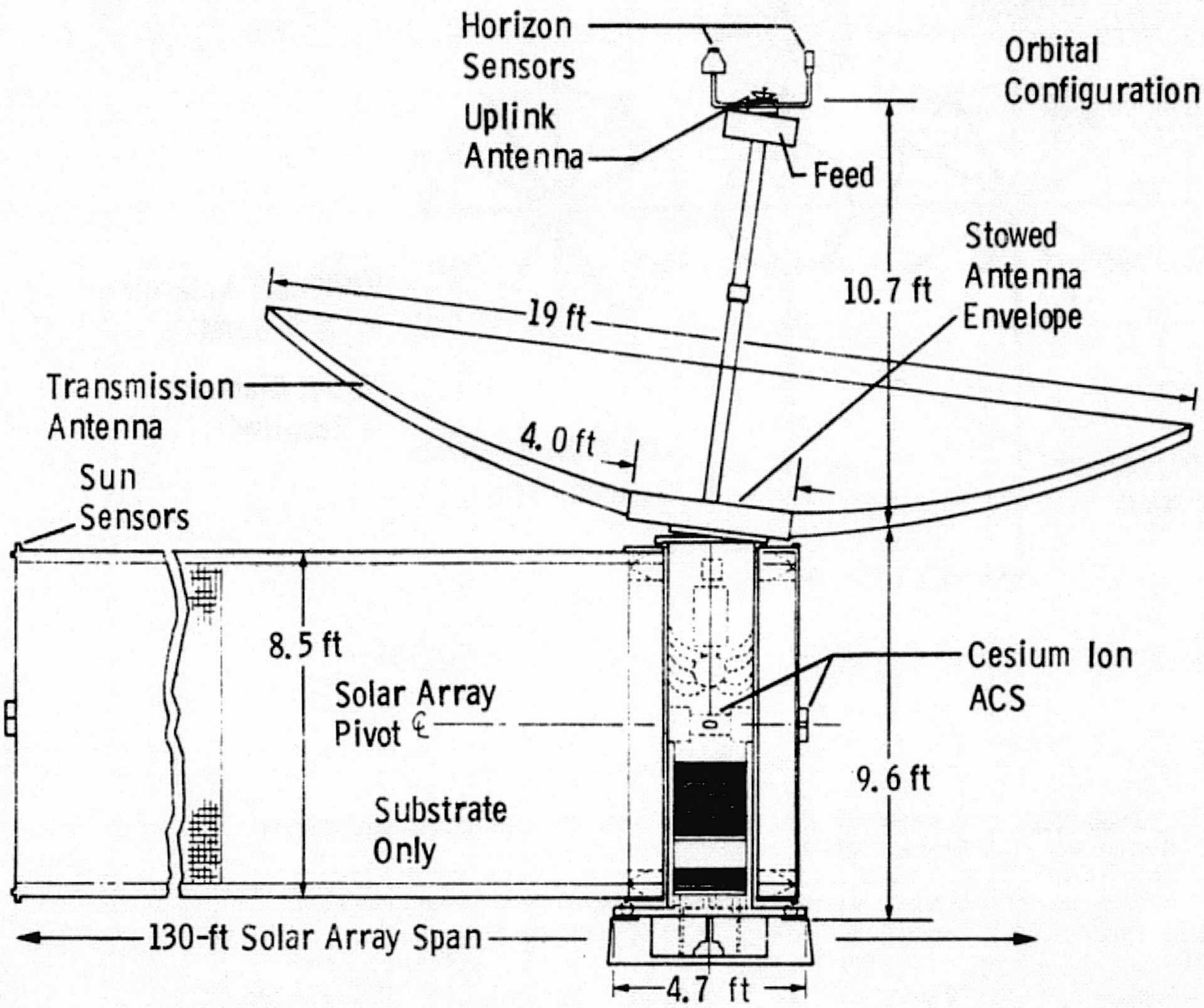


Figure IVB-1 Disaster Warning Satellite - Baseline (Non-Serviceable)

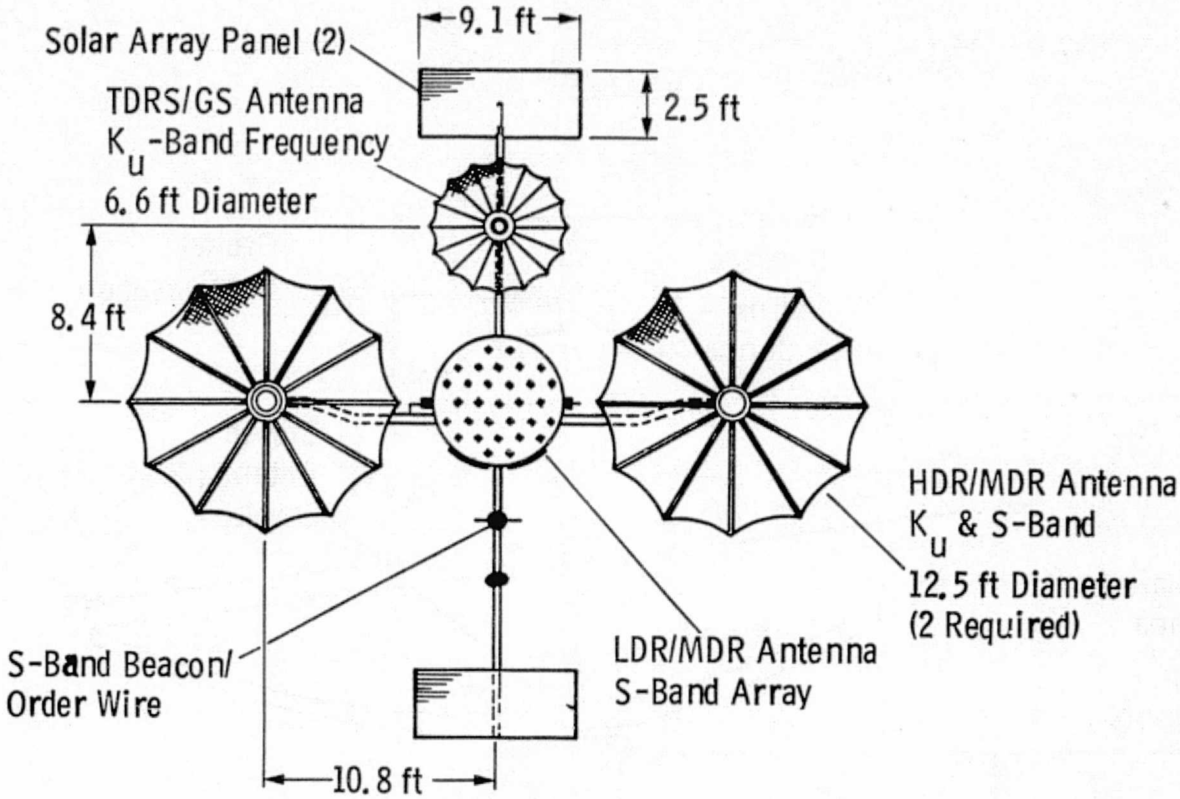


Figure IVB-2 U.S. Domsat C (TDRS) - Baseline (Non-Serviceable)

Intelsat - A view of this satellite (taken from the SSPD) is presented in Figure IVB-3.

The Intelsat will provide large capacity communications links for global commercial users and government agencies. The baseline data reported the weight to be about 3,242 lbs. Considerable mission equipment is required. Large solar arrays are used to provide the high power requirements (4400 watts). Passive thermal control methods are used. Both hydrazine and cesium ion thrusters are used for orbit attitude and position control.

Synchronous Earth Observations Satellite (SEOS) - A view (taken from the SSPD) of this satellite is presented in Figure IVB-4.

The SEOS will provide an R&D platform for multidisciplinary investigations leading to operational earth observations programs. It is estimated that the SEOS will weigh in the order of 3,300 to

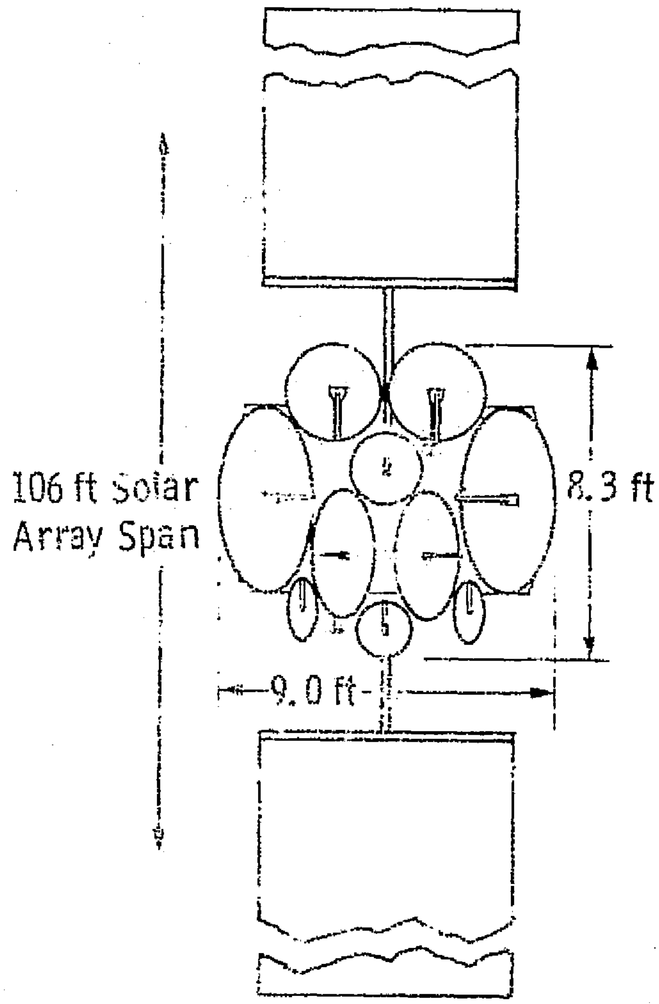


Figure IVB-3 Intelsat - Baseline (Non-Serviceable)

to 4,500 lbs. Mission equipment and supporting subsystems have nominal requirements. Electrical power is derived from small solar arrays. Thermal control is achieved by passive methods. Hydrazine thrusters are used for orbit attitude and position control. Contamination of the telescope optics may be a problem during maintenance. Telescope covers are specified in the baseline configuration.

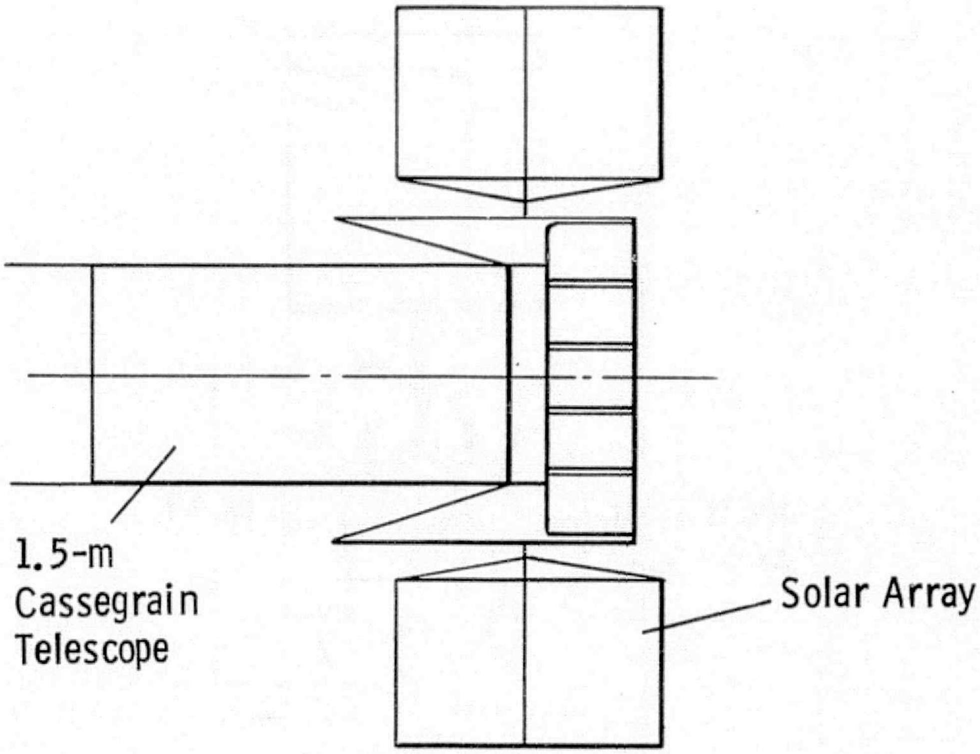


Figure IVB-4 Synchronous Earth Observations Satellite - Baseline (Non-Serviceable)

C. CONCEPTUAL DESIGNS (TASK 2)

The conceptual reconfigurations of the serviceable versions of the selected geosynchronous satellites are presented in this section. There was no baseline for the RAT. Therefore, the conceptual design of this space system included considerations for maintainable subsystems. The EOGP was originally designed for servicing. These two satellite configurations are also summarized.

The designs of these serviceable satellites were only carried far enough to provide sufficient information to enable investigating maintenance requirements. Design details were purposely limited.

1. Disaster Warning Satellite (DWS)

Views of the reconfigured serviceable DWS are presented in Figure IVC-1. The weight summary is presented in Table IVC-1.

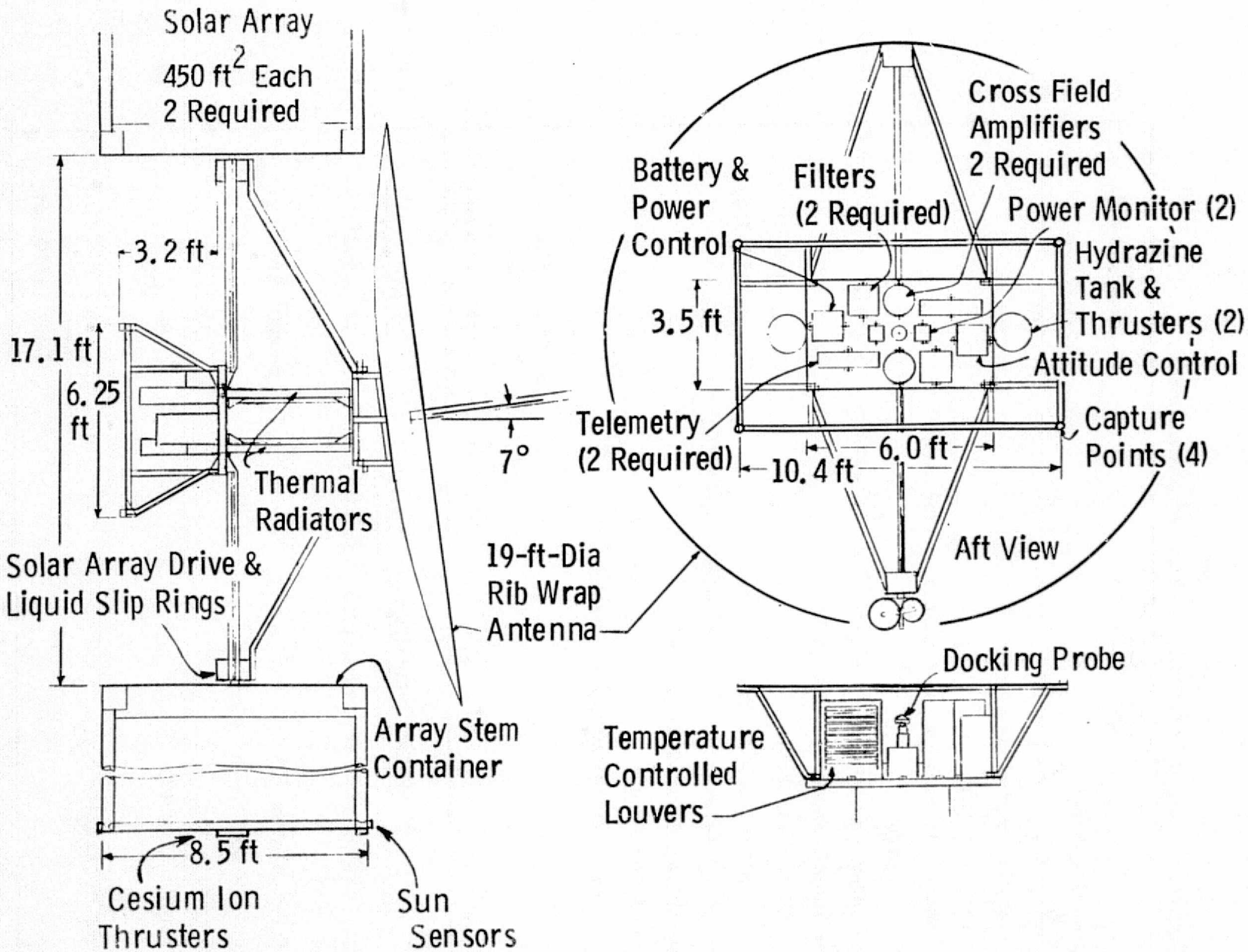


Figure IVC-1 DWS Layout - Serviceable Version

Table IVC-1 DWS Serviceable Version Weight Summary

SUBSYSTEMS	WEIGHT (LBS)	
Parabolic Antenna	130	
Power Amplifiers, Etc.	240	
Structure, Mechanisms	250	
Thermal Control	150	
Guidance, Navigation, and Stabilization (GNS)		
Sensors	45	
Reaction Control Wheel	130	
Attitude Control Systems (ACS)		
Hydrazine Systems	200	(90 dry)
Cesium Ion Systems	83	(65 dry)
Tracking, Telemetry, and Communications (TT&C)	54	
Electrical Power System (EPS)		
Solar Arrays	443	
Power Distribution and Control	179	
	1904	Wet
	1776	Dry
	128	Consumables

REPLACEABLE UNITS	WEIGHT (LBS)	
Solar Arrays with Cesium Ion Tip Thrusters (2)	526	(508 dry)
TT&C (2)	54	
ACS Propulsion (2)	200	(90 dry)
Reaction Control Wheel (1)	130	
EPS Module (1)	179	
Power Monitor (2)	40	
Filters (2)	20	
Power Amplifiers (2)	180	
	1329	(1201 dry)

The supporting systems equipment was rearranged as shown to enable replacement of the modules by a servicer docked at the anti-earth face of the DWS. This servicing configuration was taken from the Unmanned Orbital Platform Definition Study, SD73-SA-0122, September 1973.

A reaction control wheel has been added for attitude stabilization. Also, because of greater inertia, the body cesium ion thrusters in the baseline were replaced by hydrazine thrusters, which have greater thrust. The cesium ion thrusters at the array tips were retained. It was necessary to move the solar arrays storage canisters outboard to clear a docked Tug. This also required separate drive motors. The solar array, liquid metal slip ring assembly, drive motor, tip thrusters, and sun sensors on each side are considered to be replaceable as a unit. The antenna assembly, being mainly passive equipment, is assumed to not be replaceable.

2. Intelsat

The Intelsat will provide large-capacity communications links for global commercial users and government agencies. Potential serviceable configurations and equipment requirements for the Intelsat were discussed by the Communications Satellite Corporation, COMSAT Laboratories at the Second Quarterly Review at MSFC (January 1975) for the Integrated Orbital Servicing and Payloads Study (Contract NAS8-30849). Use of information from that review, supplemented by data from the DSP Space Servicing Study, Vol. III, Supporting Studies, TOR-0073(3421-07)-1, The Aerospace Corporation, August 1973, resulted in the hardware requirements presented in Table IVC-2.

These requirements differ considerably from those in the Space Shuttle Payloads Descriptions (SSPD) documents. The weight of this configuration will be 2,740 lbs. Replaceable modules total 2,016 lbs, with the largest modules being the solar array assemblies and the propulsion modules at 135 lbs each.

Views of the reconfigured Intelsat are presented in Figure IVC-2. The eight transponders are high heat producers, requiring 3.12 ft^2 each of radiator area on a north or south face to dissipate the heat. This required the transponder module shapes presented. Other satellite configurations might be more compatible with the requirements of the Intelsat, however this satellite shape and module installation methods were chosen to conform with plans in the OAM study to investigate servicing of satellites

Table IVC-2 Intelsat Serviceable Version Weight Summary

Item	Quantity	Unit Weight (lbs)	Replaceable Module Weight (lbs)	Total
6 GHz receiving antenna	1	20		20
4 GHz transmitting antenna	2	55		110
11 GHz receiving antenna	1	35		35
14 GHz transmitting antenna	1	35		35
Global coverage horns	2	9		18
Solar array assembly	2		135	270
Panels (72 ft² each)		64		
Boom and deployment mechanism		30		
Drive assembly		26		
Baseplate and mechanism		15		
Transponder	8		75	600
Equipment		60		
Baseplate		15		
Receiver	2		48	96
Equipment		33		
Baseplate		15		
Telemetry and Communications	2		40	80
Equipment		25		
Baseplate		15		
Attitude Control	2		75	150
Equipment		60		
Baseplate		15		
Battery	2		65	130
Equipment		50		
Baseplate		15		
Battery and Converter	2		75	150
Equipment		60		
Baseplate		15		
Propulsion	4		135	540
Equipment		120		
Baseplate		15		
Structure				506
Basic		224		
Wiring		12		
Module Tracks	24	220		
Docking Frame		50		
TOTAL				2,740

with different configurations. The configuration chosen is based on the satellite configuration and module installation methods proposed by The Aerospace Corporation in the DSP studies and the Operations Analysis (Study 2.1), Payload Designs for Space Servicing, ATR-74(7341)-3.

The proposed solar array assembly is a retractable FRUSA-type array. For launch in the Orbiter payload (P/L) bay, the

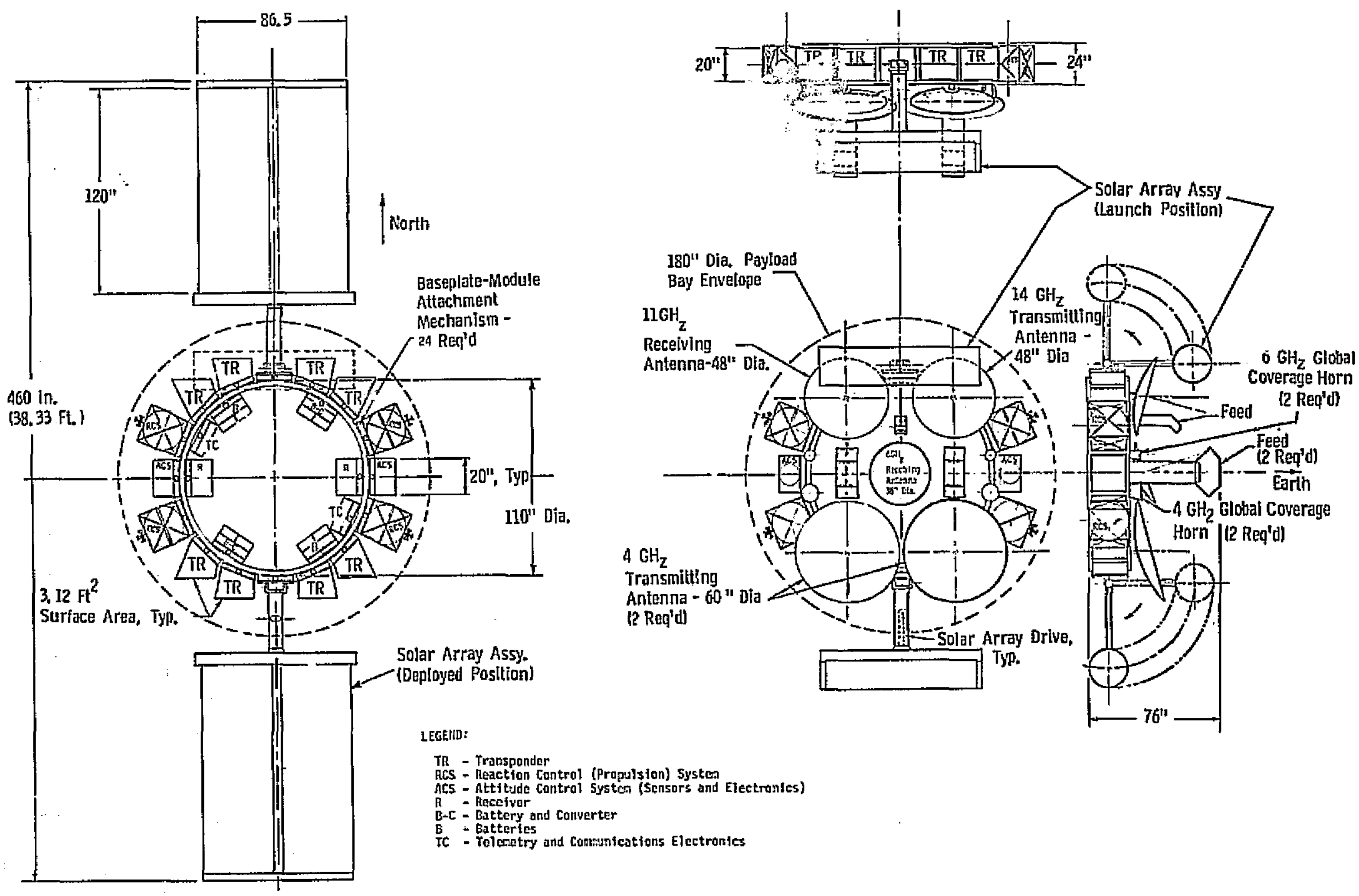


Figure IVC-2 Intelsat Serviceable Configuration

solar arrays will be hinged near the module drive assembly. The entire solar array assembly is assumed replaceable. No need is foreseen for refolding the antenna at the base hinge for in-orbit maintenance. However, the arrays would need to be folded to a launch configuration should the satellite need to be returned to earth for refurbishment.

The antennas and horns on the earth-pointing face are considered passive hardware and not replaceable. Earth pointing sensors are integral parts of the attitude control modules.

3. Synchronous Earth Observations Satellite

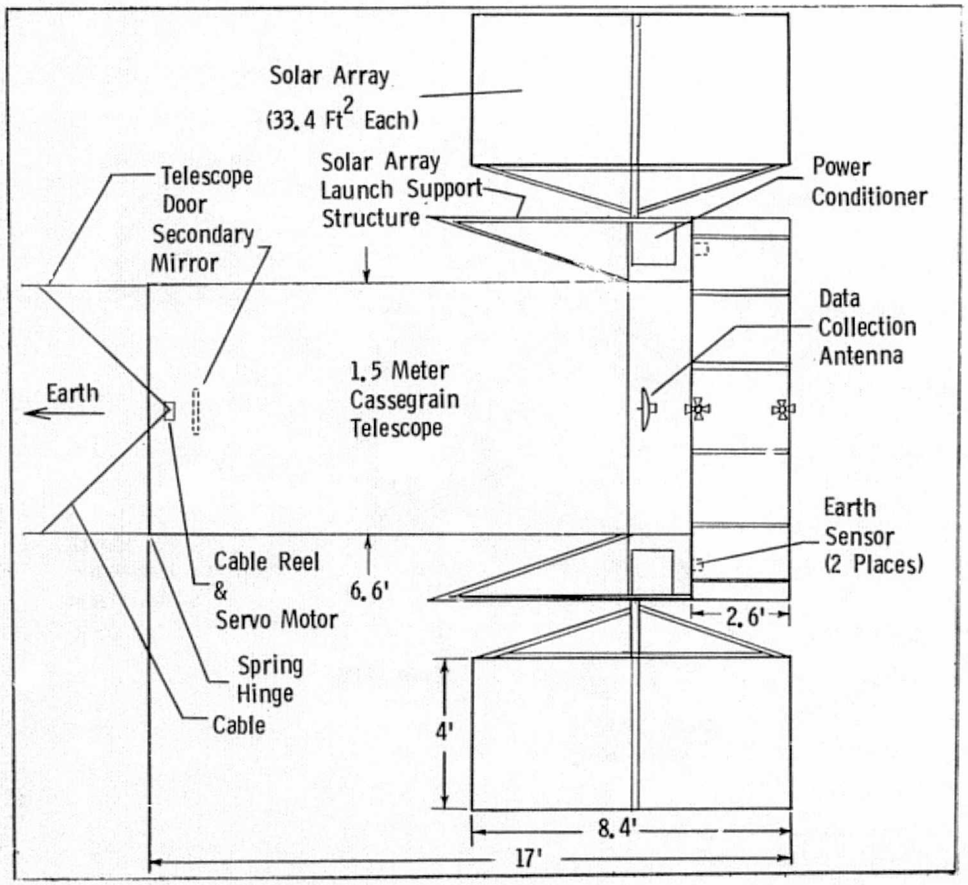
Configuration data from the SSPD on the SEOS is summarized in Figure IVC-3. Configuration data for the exploitative serviceable version of the SEOS (derived in the Study of Payload Utilization of Tug (PUT), MDC G5356, MDAC, June 1974) is summarized in Figure IVC-4.

For purposes of this study the SEOS was reconfigured, as shown in Figures IVC-5 and IVC-6, to incorporate the torroidal module arrangement with radial module extraction. This module arrangement was derived and presented in UOPD, Unmanned Orbital Platform Definition Study, SD73-SA-0122, Space Division, Rockwell International, September 15, 1973. A weight summary of the reconfigured serviceable SEOS is presented in Table IVC-3. The reconfigured SEOS combines elements from both the PUT study and the SSPD. The SSPD structure and thermal control weights were used as the structure appeared similar to the reconfigured SEOS. The mission sensors and data collection system were separated into two modules to distribute the weight and the heat dissipation.

4. U. S. Domsat C (Tracking and Data Relay Satellite)

a. Baseline Configuration - Two tracking and data relay satellites (TDRS) will operate in geosynchronous orbit to provide forward and return telecommunications links for low, medium, and high data rate satellite users in earth orbit.

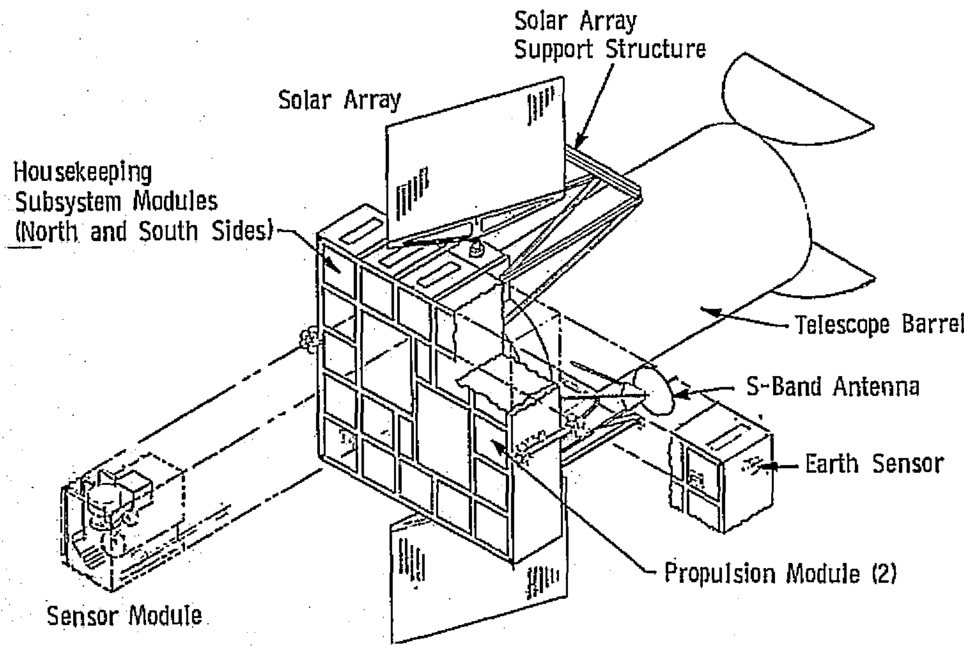
The baseline configuration presented herein is the second alternate configuration developed in the TDRS Configuration and Tradeoff Study (Part II) by Rockwell International. This configuration was chosen since it most closely matched the Level A data presented in the July 1974 SSPD. The SSPD referenced the Hughes Aircraft Co. TDRS study. However, the Hughes' study did not consider the same mission equipment as listed in the SSPD and considered



EQUIPMENT	WEIGHT (LBS)
MISSION EQUIPMENT:	
Telescope	1325
Sensor Assembly*	350
Data Collection Electronics	53
Data Collection Antenna	73
SUBSYSTEM EQUIPMENT:	
Structure, Mechanisms	450
Thermal Control	90
GNS	296
ACS	294 (147 Dry)
TT&C	78
Electrical (557 W. Bol)	328
Data Processing	38
TOTAL	3375 (3228 Dry)

*Contains: 1 Linear Silicon Diode Array
12 Photo Multipliers
9 Silicon Detectors
6 Mercury Cadmium Telluride Detectors
1 Immersed Thermistor Bolometer

Figure IVC-3 Baseline SEOS from SSPD



EQUIPMENT	UNIT WEIGHT (MODULES)		TOTAL WEIGHT (LBS)
MISSION EQUIPMENT:			
Telescope			1325
Sensors	350	(1)	350
Data Collection System	57	(1)	57
SUBSYSTEM EQUIPMENT:			
Structure and Thermal Control			640
Attitude Control			
Sensing	106	(2)	212
Momentum Storage	100	(2)	200
TT&C			
S-Band Transponder	85	(1)	85
S-Band Antenna		(Ext)	13
Data Processing	61	(1)	61
Electrical Power			
Power Conditioning and Storage	126	(2)	251
Power Generation	71	(2)	142
Solar Arrays and Shafts		(2-Ext)	82
Propulsion (Wet)	171	2	342
TOTAL		(14)	3760

Figure IVC-4 Baseline SEOS from PUT Study

spin stabilization instead of 3-axis stabilization as stated in the SSPD.

The following mission particulars and general characteristics apply to the baseline TDRS.

Launch - Shuttle to LEO, P/L-Tug deployed by RMS, checked out in LEO, P/L carried to geosynchronous orbit by Tug.

Schedule - Three satellites (includes one spare) launched in 1983 and three (includes one spare) launched in 1988.

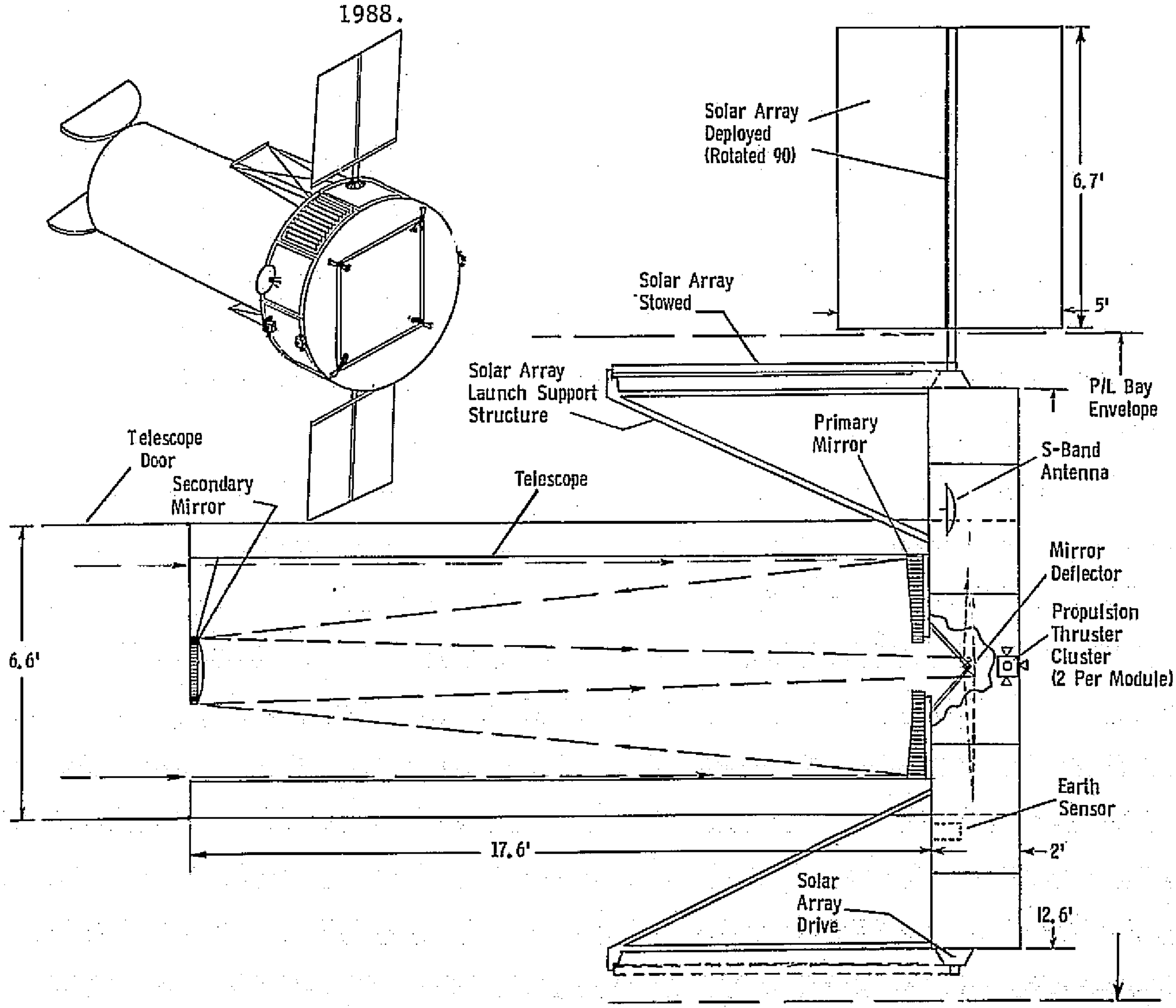


Figure IVC-5 SEOS Serviceable Version

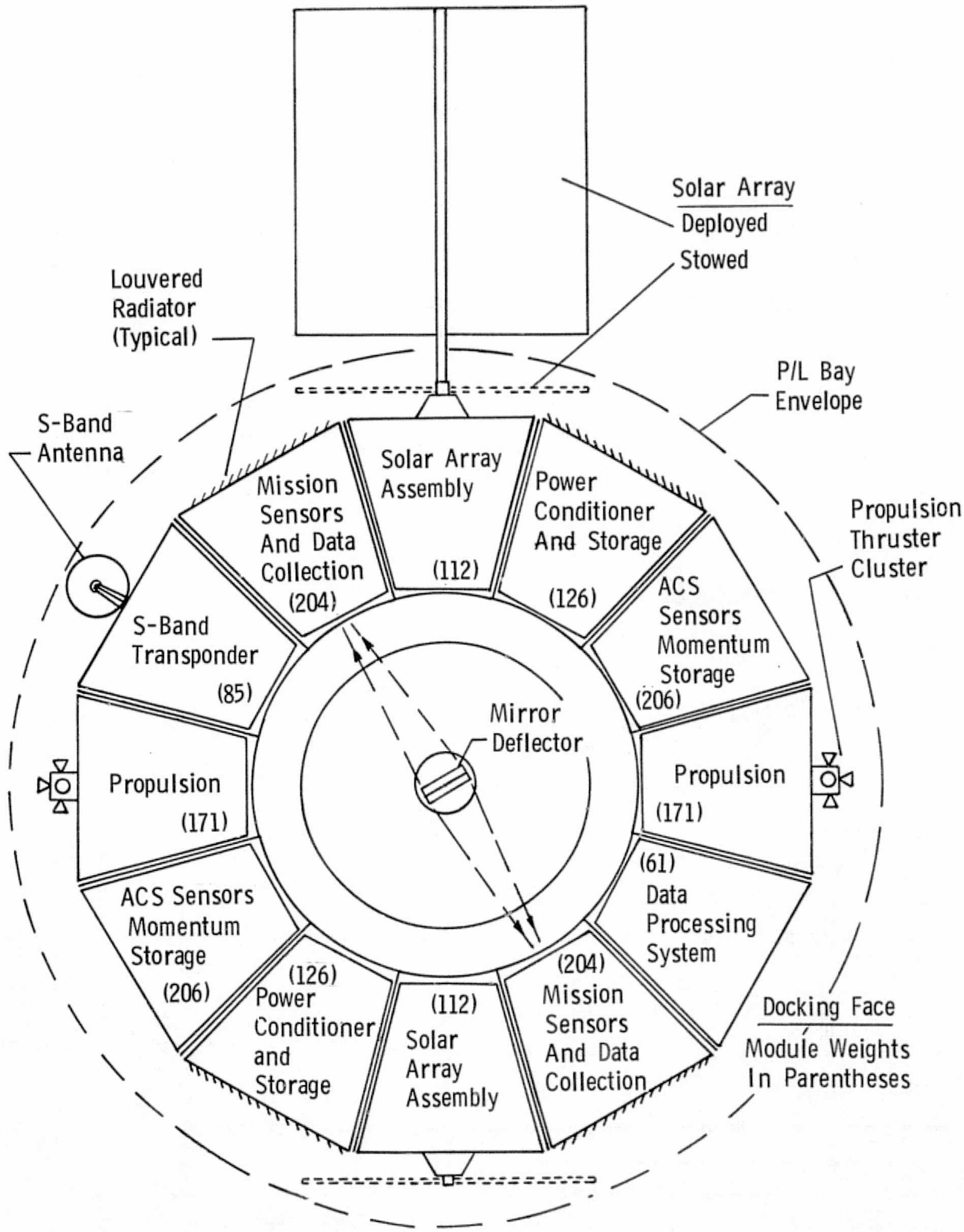


Figure IVC-6 Module Arrangement - Serviceable SEOS

Table IVC-3 SEOS Serviceable Version Weight Summary

Item	Number of Replaceable Units	Module Unit Weight (lbs)	Total Weight (lbs)
<u>Mission Equipment:</u>			
Telescope			1325
Sensors and Data Collection System	2	204	407
<u>Subsystem Equipment:</u>			
Solar Array Assembly	2	112	224
Power Conditioner and Storage Batteries	2	126	251
Attitude Control Systems Earth and Sun Sensors Momentum Wheels	2	206	412
Attitude Propulsion	2	171	342
Data Processing System	1	61	61
TT&C S-Band Transponder S-Band Antenna	1	85	85
Structure and Thermal Control			540
Docking Frame			50
TOTAL			3697

Replaceable Units:

12 modules

1784 lbs total

Inclination - 3 ± 0.1^{o}

Longitude - 41^{o}W and 171^{o}W

Design Lifetime - 7 years

Weight - 304 kg (669 lbs) launched
286 kg (630 lbs) expended

Dimensions - meters (feet):

Ascent/stowed - 5 x 2 diam (16.4 x 6.6 diam) approx.
Deployed - 10 x 11 x 3 (32.8 x 36.1 x 9.9) approx.
(See Figure IVC-7)

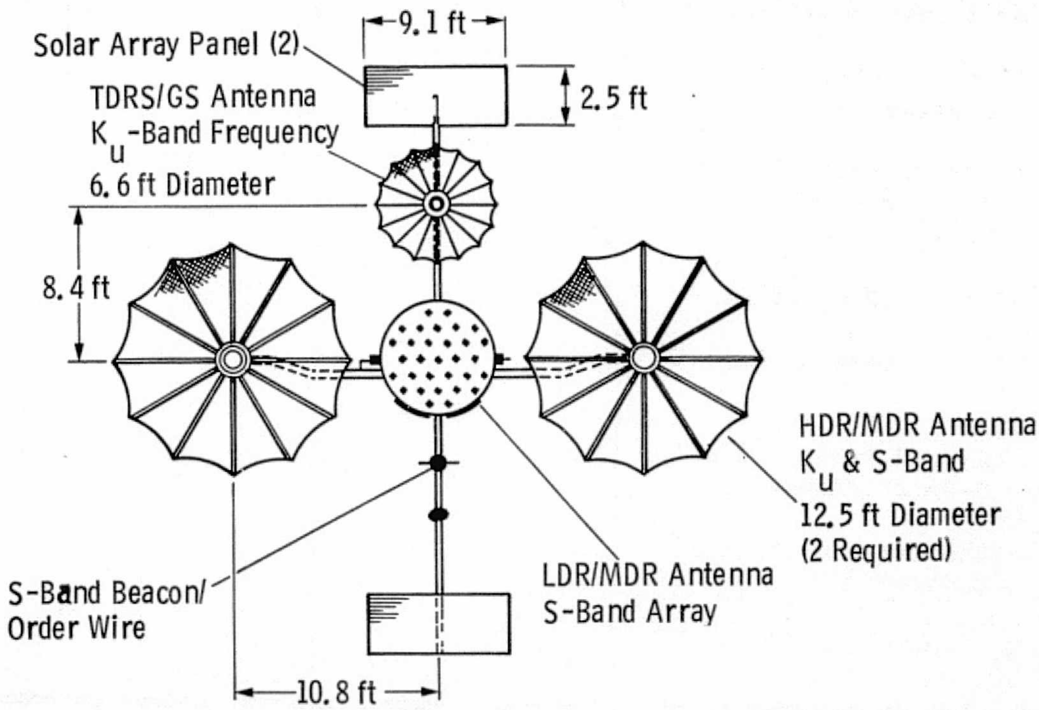


Figure IVC-7 Baseline TDRS

Mission equipment includes the following items:

- Two mechanically steerable 3.8-meter parabolic antennas for communications with satellites in the S and Ku bands at high and medium data rates (HDR/MDR).

- One mechanically steerable 2.0-meter parabolic antenna for communications with ground systems (GS) in the Ku band.

- One 31-element array antenna for communications with satellites in the S band at low and medium data rates (LDR/MDR).

- Eight TT&C backup omni antennas.

- One S-band beacon/order wire.

Each of the 3.8-meter parabolic reflectors is formed by 12 rigid ribs of 1.5-inch diameter thin-wall aluminum tubes which support and contour the elastic mesh surface. The mesh is constructed from 7-strand bundles of 0.7 mil Chrome I-R wire knitted into a wire screen. The mesh is plated with electroless nickel, gold, and vapor-deposited aluminum. The reflector ribs are restrained in the stowed configuration by a moment-resisting joint with a preload maintained by a tensioned cable around the rib tips. On deployment command, a redundant set of guillotine cutters servers the cable. Deployment is accomplished by redundant energy drive systems rotating a ball screw within a recirculating ball nut. The resultant linear motion of the ball nut rotates each rib through an indivdual linkage to each rib. The primary drive is a 5-inch constant torque spring motor. A backup drive system consists of two miniature torque motors driven through a 60:1 ratio gear system. Latching in the deployed condition is accomplished by driving the ball nut carrier and linkages through an over-center. A reverse torque of 8 inch-pounds on the ball-screw is required to back drive the mechanism through the latching toggle action. The antenna can be remotely stowed during ground testing by reversing the current to the electric motors. The deployment control unit, located at the base of the feed support cone, sequences and controls the deployment and provides telemetry to indicate deployment initiation, progress and completion.

The 2-meter TDRS/GS antenna is of rigid construction and is mounted to one solar array strut. The entire assembly is rotated 90° for deployment.

The S-band array consists of 31 single-helix elements (29 receiving and 2 transmitting). Each element is a thin wall dielectric material tube supporting a conductive material tape wound in a helix on the tube outer surface. The elements are assembled from the rear through holes in the face of the equipment housing.

The TT&C backup antennas are VHF omni whips. One set of four whips located radially around the rear of the equipment housing is utilized during launch when the primary antennas are stowed. After on-orbit deployment of the primary antennas, a TT&C backup to the TDRS/GS Ku-band link is supplied by another set of omni-whip antennas mounted around the rim of the S-band array antenna.

The S-band beacon/order wire is attached to the lower solar array strut and is deployed into position as the strut is deployed. The antenna is a 2-inch diameter helix mounted on an 11-inch ground plane.

The weight summary of the mission equipment is presented in Table IVC-4.

The spacecraft body (see Figure IVC-8) consists of an inner aluminum tapered cone, a transverse equipment shelf of aluminum honeycomb and the outer body shell of aluminum honeycomb that closes off and protects the internal equipment and houses the thermal louvers. The center cone-shaped void was where the apogee motor was installed for the configuration launched with a Delta booster.

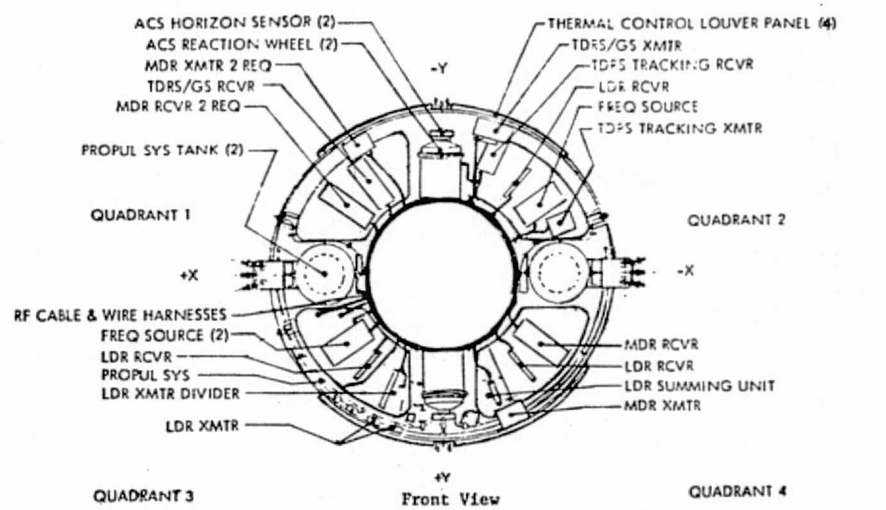
The equipment shelf is an aluminum honeycomb bulkhead (1.5 inch thick with 0.010 inch face sheets) that provides the primary equipment mounting surface. Insert panels bolted to the main bulkhead are also used for mounting equipment. The structural weight is 38.04 kg.

Electrical power is provided by solar arrays and by nickel-cadmium batteries during eclipse. The solar arrays are deployed on-orbit and are continually rotated to remain normal to the sun. The arrays are curved for better packaging during launch. Electrical system weights are:

Solar Array		26.1 kg
Panels (2)	15.1	
Drive mechanism (2)	6.8	
Linkage & fittings (2)	4.2	
Power Conditioning & Distribution		22.1 kg
Charge & discharge	5.1	
Central control & logic	2.3	
Packaging	2.2	
Shunt dissipators	1.1	

Table IVC-4 Baseline TDRS Mission Equipment Weight Summary

		Weight (kg)
HDR/MDR System		
No. 1 receiver		4.5
No. 2 receiver		4.5
No. 1 transmitter		6.4
No. 2 transmitter		6.4
Antenna (2)		34.7
Reflector	7.09	
S-band feed	1.04	
Ku-band feed	.95	
Control/Elec.	2.26	
Gimbal	2.26	
Rotary joints	.95	
Support strut	2.81	
	17.36	
TDRS/GS System		
Transmitter		9.6
Receiver		2.2
Antenna		7.5
Reflector	2.44	
Ku-band feed	.95	
Gimbal	1.47	
Control/Elec.	2.27	
Rotary joints	.39	
	7.52	
S-Band Array System		
Elements (31)		1.32
Receivers (29)		13.15
Transmitters (2)		6.40
FDM module		.64
Local frequency reference		6.19
TT&C Omni Antennas (8)		8.6
S-Band Beacon/Order Wire		
Transceiver		2.5
Antenna Helix		.1
Miscellaneous		
DC wiring		6.0
RF cabling and W/G		6.0
Frequency source		3.5
		130.2 kg



NOTE: The RI study did not update the equipment layout for the configuration incorporating the S-band array.

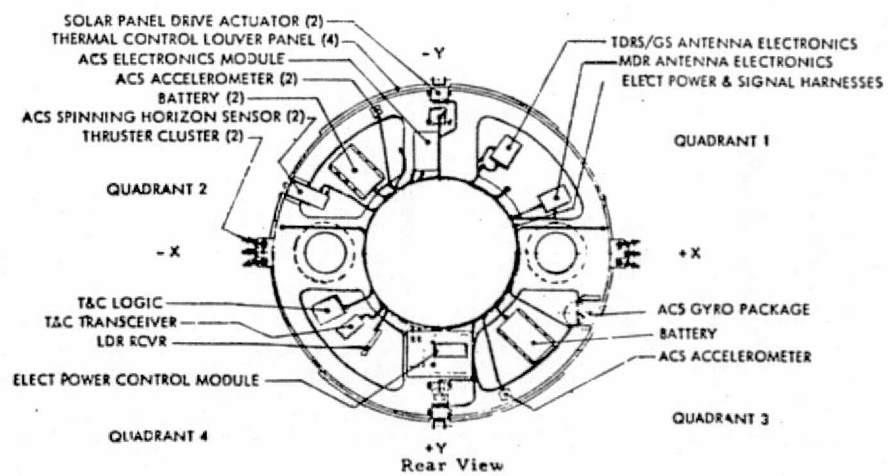


Figure IVC-8 Baseline TDRS Equipment Layout

ORIGINAL PAGE IS OF POOR QUALITY

Power conditioner voltage	2.3	
Cabling	9.1	
Energy Storage		20.1 kg
Batteries (2)		

The available solar array power (watts) is:

	Equinox	Solstice
Beginning of Life	487	436
End of Life	417	375

Power demands vary from 307 to 381 watts. If the power demand exceeds that available, or is insufficient to charge batteries in a reasonable time, some telecommunications service will be temporarily reduced. Two 16-cell, 12-amp-hour batteries will supply the power during eclipse. They have a capacity of 460 watt-hours. Limiting maximum depth of discharge to 60% results in 276 watt-hours of usable energy.

Since all antenna beams are steerable, pointing accuracy requirements imposed by the telecommunications are not severe. Of greater importance is the need for accurate knowledge of the spacecraft attitude to establish a reference for pointing the antennas for S-band. Spacecraft attitude is maintained by momentum bias/momentum transfer three-axis stabilization.

Attitude Determination Accuracy (Knowledge) -

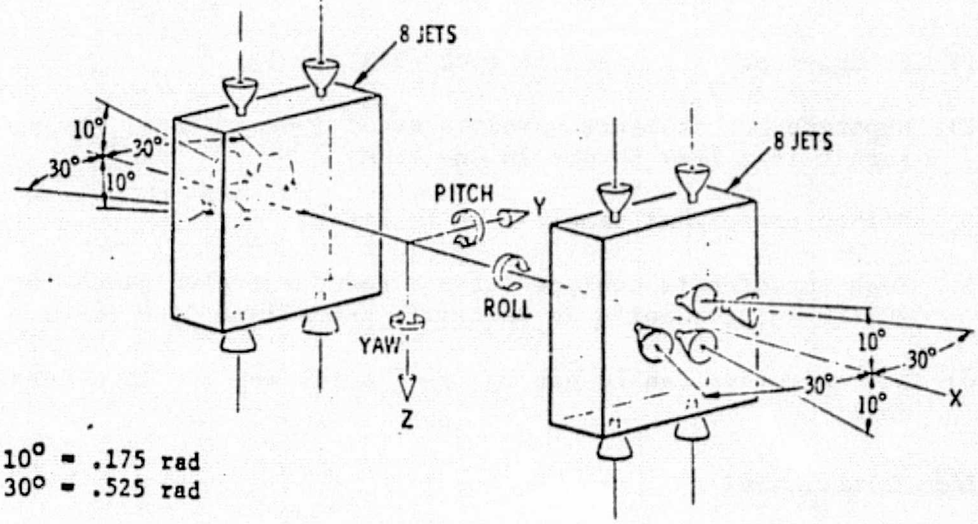
Roll - 0.25°

Pitch - 0.25°

Yaw - 0.25°

Spacecraft Attitude Pointing Accuracy - ± 0.58°
Attitude Stabilization System Weight - 26.2 kg
Stationkeeping Accuracy - ± 0.125° (corrections approx. every 17 days)
Momentum Dumping Maneuvers - No more than once per day
Longitudinal Station Change - One of 65°, every 20 days

The reaction control system includes two jet thruster quads of eight jets each and two N_2H_4 propellant tanks with GN_2 pressurant (see Figure IVC-9). Initial thrust levels of 0.27 lb and final thrust levels of 0.09 lb are predicted. Key features of the RCS are:



10° = .175 rad
30° = .525 rad

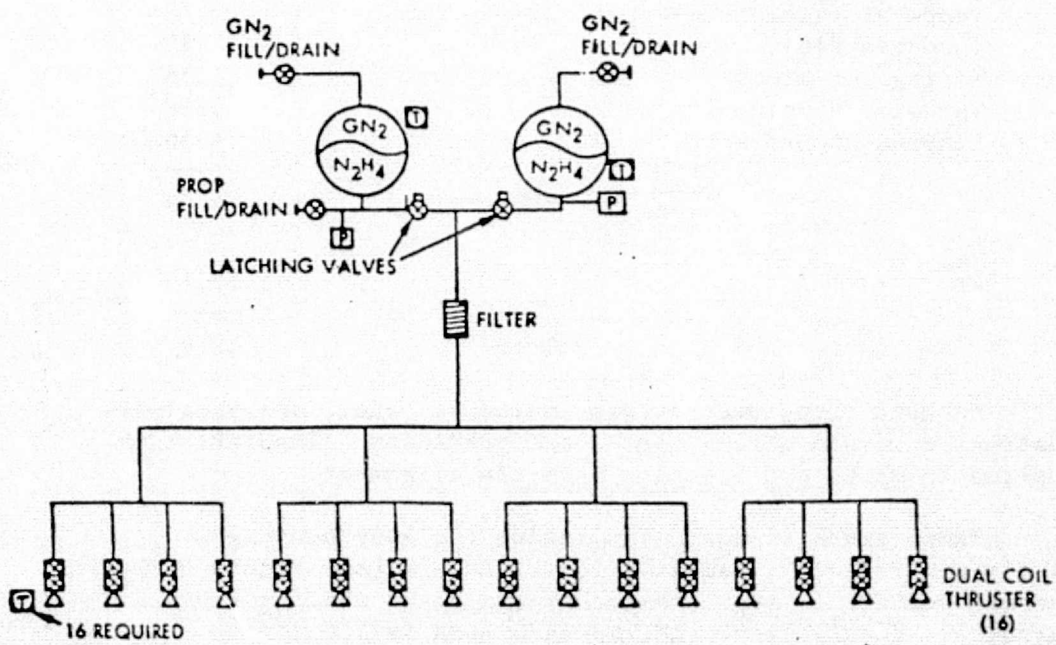


Figure IVC-9 Baseline TDRS RCS Configuration

1) It operates in a blowdown mode (3.35 to 1).

2) Separate GN_2 fill/drain valves avoid loss of total pressurant if a leak occurs in one tank.

3) Either propellant tank can be isolated.

4) Each thruster is equipped with a redundant (two seats in series and two coils in parallel) propellant flow valve.

5) All maneuvers can be accomplished after any two thrusters fail.

<u>RCS weights (kg)</u>

GN_2 fill/drain (2)		0.27
Propellant tank (2)		4.99
Latching valve (2)		0.54
Propellant fill/drain		0.14
Filter, 15μ abs		0.18
Pressure transducer (2)		0.27
Temperature transducer, tanks (2)		0.27
Temperature transducer, thrusters (16)		0.18
Thruster (16)		4.35
Wiring and lines		1.36
Thruster housing (2)		0.68
Trapped propellant		1.36
	Subtotal	14.59
GN_2		0.27
Propellant		17.55
	Total	32.4

Exposed structure, arrays, antennas, etc., are passively controlled within design limits and gradients. Insulation is applied to masts and feeds to maintain alignment.

Temperature of equipment inside the body housing is primarily controlled by radiator/louver panels (see Figure IVC-8). Thermal control is supplemented by radiation windows (insulation cutouts). Each louver panel in each quad is 3.6 ft^2 in area and is capable of 44.4 watts heat rejection for a panel temperature of 30^oC when the solstice solar vector lies in the XY plane. Based on the power dissipation loads presented in Table IVC-5, the additional heat dissipation required for the worst case quad

is 57.7 watts. The radiation window size required is 1.44 ft^2 with no solar incidence. This size is assumed for each quad. The high-heat-source transmitters are located directly at the radiator panels. Thermal control system weights total 8.73 kg.

The total baseline TDRS weight summary is presented in Table IVC-6.

The Rockwell International configuration of the TDRS incorporating the S-band array was discussed in the TDRS Configuration and Tradeoff Study (Part II) to considerable depth. However, the layout of the mission equipment and supporting subsystems was not presented. Therefore, this reconfiguring was done as part of the present maintenance study to derive a baseline configuration for the TDRS. Results are presented in Table IVC-7.

b. Serviceable Configuration - The TDRS was reconfigured to a serviceable version. The front view of the on-orbit configuration is presented in Figure IVC-10. The Shuttle launch configuration is shown in Figures IVC-11 and IVC-12. Figure IVC-12 also presents the layout of the supporting subsystem modules. The module weights and power requirements are detailed in Table IVC-8. All TDRS weights are summarized in Table IVC-9. Replaceable units and their weights are itemized in Table IVC-10.

The following considerations were used in the derivation of the serviceable configuration or were an outgrowth of this configuration.

1) The stowed envelope should be compatible with the Shuttle P/L bay. It was desirable to be able to stack 2 or 3 TDRSs, with a baseline Tug, in a single P/L bay volume.

2) The high heat producing subsystems should be located on north-south faces for more effective heat dissipation.

3) The rectangular-matrix module arrangement presented in the PUT study for the CSCSAT was selected.

4) The arrangement of equipment on the TDRS should be balanced as much as possible for best CG location and to reduce solar pressure and gravity gradient torques.

5) A docking frame, umbilical connector, laser radar reflectors, and capture mechanisms were added. Docking points on a 14-foot diameter were chosen to clear the 6-ft by 10-ft module matrix.

Table IVC-5 Baseline TDRS Power Dissipation (Watts)

Quad 1.		Quad 2.	
Front side		**Front side**	
Horizon sensor	0.25	Horizon sensor	0.25
Reaction wheel	3.00	Reaction wheel	3.00
MDR XMTR	52.00	TDRS/GS XMTR	40.60
TDRS/GS RCVR	5.30	TDRS Track RCVR	5.30
MDR RCVR	8.20	LDR RCVR	9.10
Back side		Freq. source	8.00
Solar panel drive	1.63	TDRS Track XMTR	2.00
MDR electronics	4.00	**Back side**	
	Σ 74.38	Solar panel drive	1.63
		ACS electronic	3.00
			Σ 72.88

Quad 3.		Quad 4.	
Front side		**Front side**	
Freq source	8.00	MDR RCVR	8.20
LDR RCVR	9.10	LDR RCVR	9.10
LDR XMTR	66.00	MDR XMTR	52.00
Horizon sensor	0.25	Horizon sensor	0.25
Reaction wheel	3.00	Reaction wheel	3.00
Back side		**Back side**	
ACS gyro	2.00	T&C	10.50
Pwr module	10.00	LDR RCVR	9.10
	Σ 98.35	Pwr module	10.00
			Σ 102.15

NOTE:
The RI study did not update this table for the configuration incorporating the S-band array.

Table IVC-6 Baseline TDRS Weight Summary

	Weight (kg)
Mission Equipment	
HDR/MDR system	56.5
TDRS/GS system	19.3
S-band array	27.7
TT&C omni antennas	8.6
S-band beacon order wire	2.6
Miscellaneous wiring, cabling, etc.	15.5
Structure	38.0
Electrical Power	
Solar array	26.1
Power conditioning & distribution	22.1
Batteries	20.1
Attitude Stabilization and Control	
Stabilization	26.2
Reaction control	14.6
Thermal Control	8.7
	286.0 Expended Weight
Propellant and GN_2	17.8
Total	303.8 kg Launch Weight (669 lbs)

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Table IVC-7 Baseline TDRS Equipment Summary

Item	Weight (kg)	Weight (lbs)	Power Dissipation (Watts)
Quad 1 - Front			
ACS horizon sensor	2.5	5.5	0.5
ACS reaction wheel	4.6	10.1	6.0
LDR transmitter	3.2	7.1	66.0 (one at a time)
TDRS/CS receiver	2.2	4.8	5.3
MDR/HDR receiver	4.5	9.9	8.2
Propulsion system tank/hdw	4.0	8.8	
DC wiring	0.7	1.5	
RF cabling	0.8	1.8	
ACS wiring	0.3	0.7	
RCS wiring	0.1	0.2	
LDR receivers (7 distributed)	3.2	7.0	9.0
GN_2	0.1	0.2	
Propellant	8.8	19.6	
	35.0	77.0	95.8
Quad 1 - Rear			
TDRS/CS antenna electronics	2.3	5.1	4.0
MDR/HDR antenna electronics	4.6	10.1	4.0
DC wiring	0.7	1.5	
RF cabling	0.8	1.8	
Power cabling	2.2	4.8	
ACS wiring	0.3	0.7	
RCS wiring	0.1	0.2	
	11.0	24.2	8.0
Quad 1 total	46.0 kg	101.2 lbs	103.8 watts
Quad 2 - Front			
TDRS/CS transmitter	9.6	21.1	40.6
Frequency source	1.8	4.0	8.0
DC wiring	0.7	1.5	
RF cabling	0.8	1.8	
MDR/HDR transmitter	6.4	14.1	52.0
ACS wiring	0.3	0.7	
RCS wiring	0.1	0.2	
LDR receivers (8 distributed)	3.6	7.9	11.2
	23.3	51.3	111.8
Quad 2 - Rear			
Solar panel drive actuator	3.4	7.5	1.6
ACS electronics module	2.5	5.5	3.0
ACS accelerometer	1.0	2.2	0.5
Battery	10.0	22.0	
Thruster cluster	4.3	9.5	
DC wiring	0.7	1.5	
RF cabling	0.8	1.8	
Power cabling	2.2	4.8	
ACS wiring	0.3	0.7	
RCS wiring	0.1	0.2	
	25.3	55.8	5.1
Quad 2 total	48.6 kg	106.9 lbs	116.9
Quad 3 - Front			
Frequency source	1.8	4.0	8.0
Propulsion system piping	0.2	0.4	
DC wiring	0.7	1.5	
RF cabling	0.8	1.8	
MDR/HDR transmitter	6.4	14.1	52.0
MDR/HDR receiver	4.5	9.9	8.2
ACS wiring	0.3	0.7	
RCS wiring	0.1	0.2	
LDR receivers (7 distributed)	3.2	7.0	9.8
S-band beacon transceiver	2.5	5.5	2.0
	17.5	38.5	80.0
Quad 3 - Rear			
Thruster cluster	4.0	8.8	
ACS gyro package	2.6	5.7	2.0
Battery	10.0	22.0	
ACS accelerometer	1.0	2.2	0.5
Solar array drive actuator	3.4	7.5	1.6
DC wiring	0.7	1.5	
RF cabling	0.8	1.8	
Power cabling	2.2	4.8	
ACS wiring	0.3	0.7	
RCS wiring	0.1	0.2	
	25.1	55.3	4.1
Quad 3 total	42.6 kg	93.0 lbs	84.1 watts
Quad 4 - Front			
Propulsion system tank/hdw/piping	4.5	9.9	
Local frequency reference	6.2	13.6	8.0
FDM module	0.6	1.3	10.0
LDR transmitter	3.2	7.1	66.0 (only one at a time)
ACS horizon sensor	2.5	5.5	0.5
ACS reaction wheel	4.6	10.1	6.0
DC wiring	0.7	1.5	
RF cabling	0.8	1.8	
ACS wiring	0.3	0.7	
RCS wiring	0.1	0.2	
LDR receivers (7 distributed)	3.2	7.1	9.8
GN_2	0.1	0.2	
Propellant	8.8	19.4	
	35.6	78.4	100.3
Quad 4 - Rear			
TT&C processor	4.4	9.7	2.0
TT&C transceiver	1.8	4.0	8.5
Electrical power control module	13.0	28.6	20.0
DC wiring	0.7	1.5	
RF cabling	0.8	1.8	
Power cabling	2.2	4.8	
ACS wiring	0.3	0.7	
RCS wiring	0.1	0.2	
	23.3	51.3	30.5
Quad 4 Total	58.9 kg	129.2 lbs	130.8 watts
External			
MDR/HDR antennas (2)	30.2	66.5	
TDRS/CS antenna	5.2	1.6	
S-band array elements (31)	1.3	2.9	
TT&C Omni Antennas (8)	2.4	5.3	
S-band beacon/order wire antenna	0.1	0.2	
Solar arrays (2)	19.3	42.5	
Solar aspect sensors (2)	3.0	6.6	
	61.5	135.0	
Structure			
Equipment housing	38.0	83.6	
Louver panels	8.7	19.1	
	46.7	102.8	
Vehicle total	305.3 kg	671.0 lbs	359.6 watts (with one LDR transmitter operating)

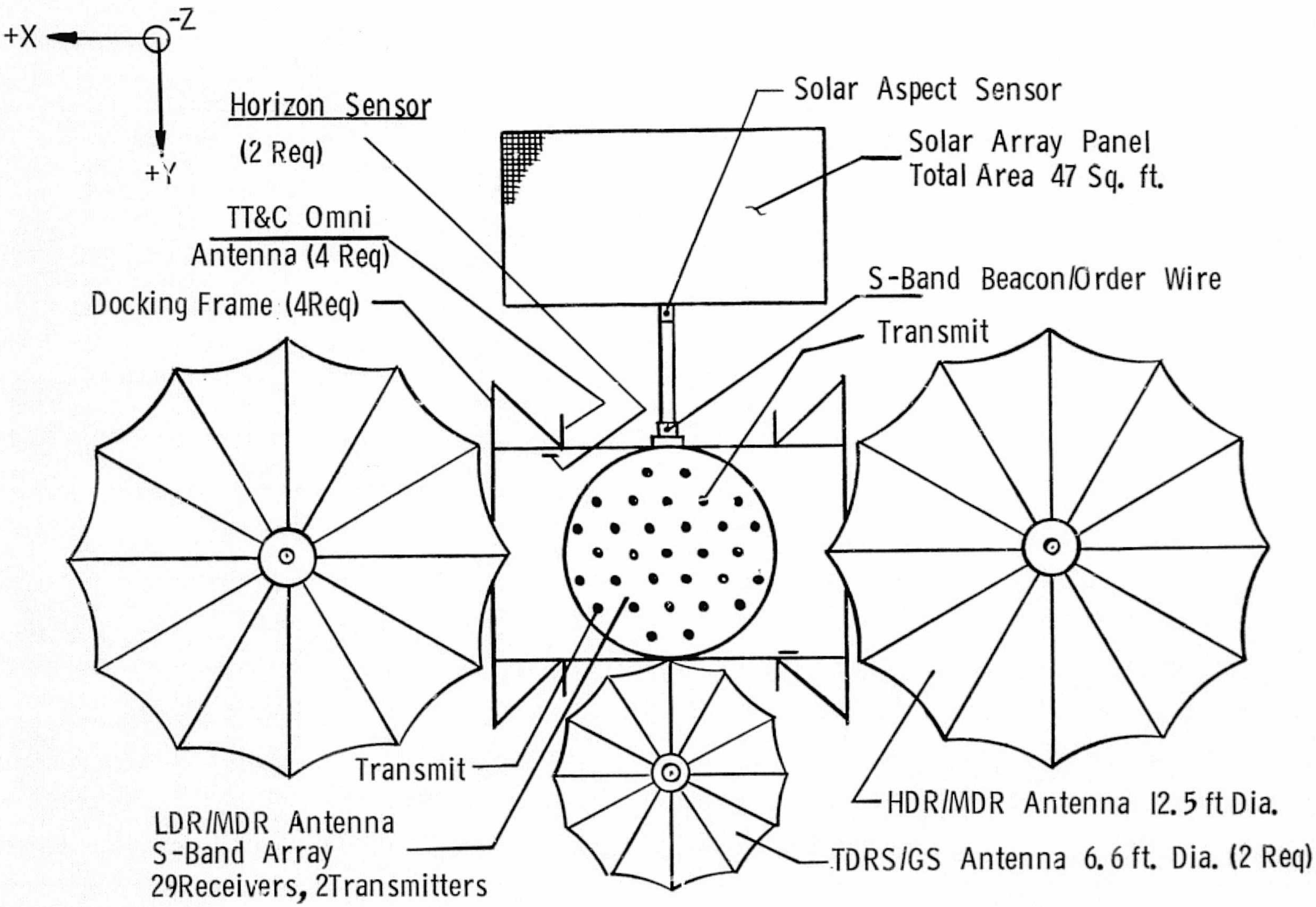

Figure IVC-10 Front Layout - Serviceable TDRS

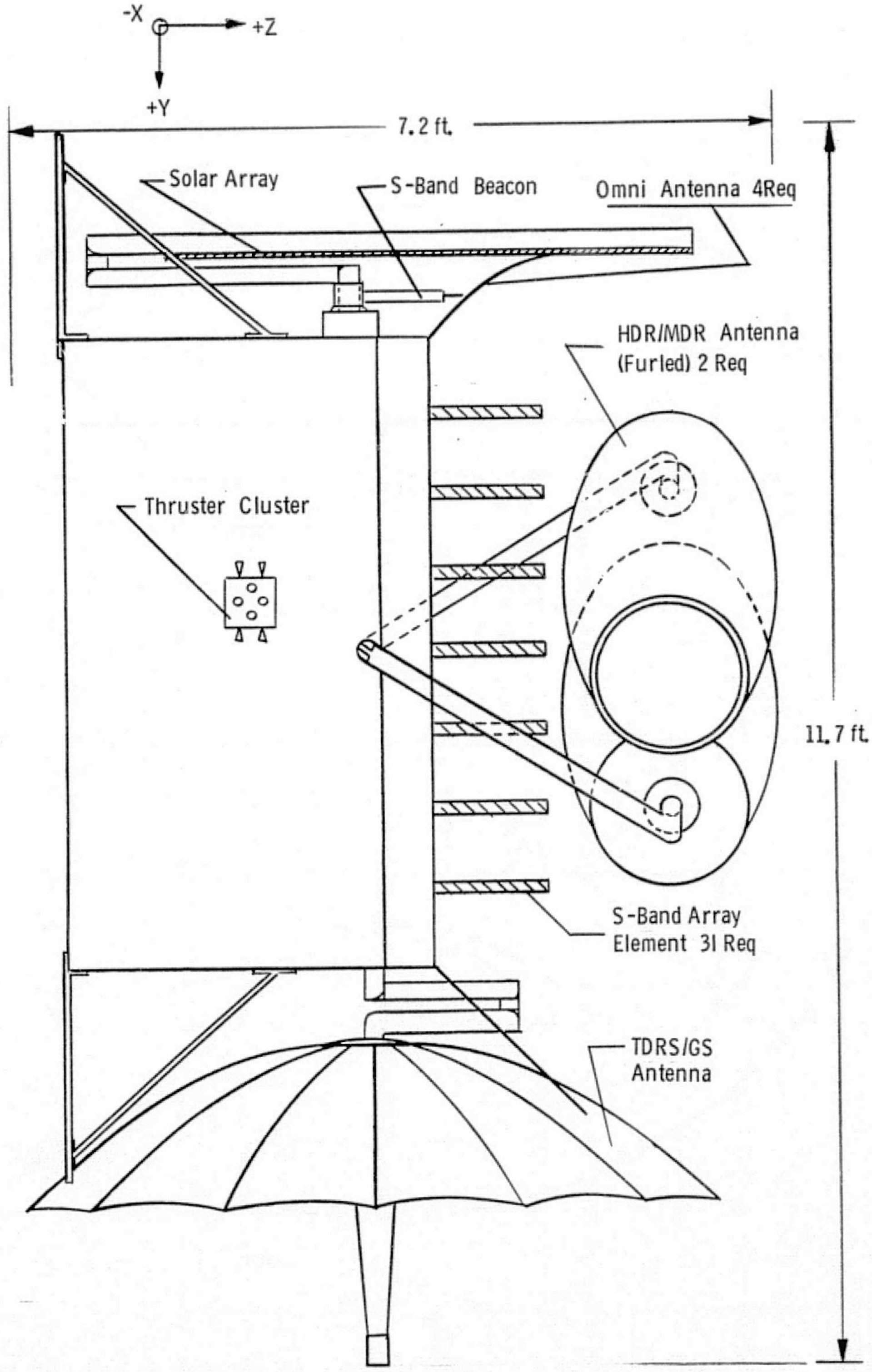

Figure IVC-11 Stowed Configuration - Serviceable TDRS

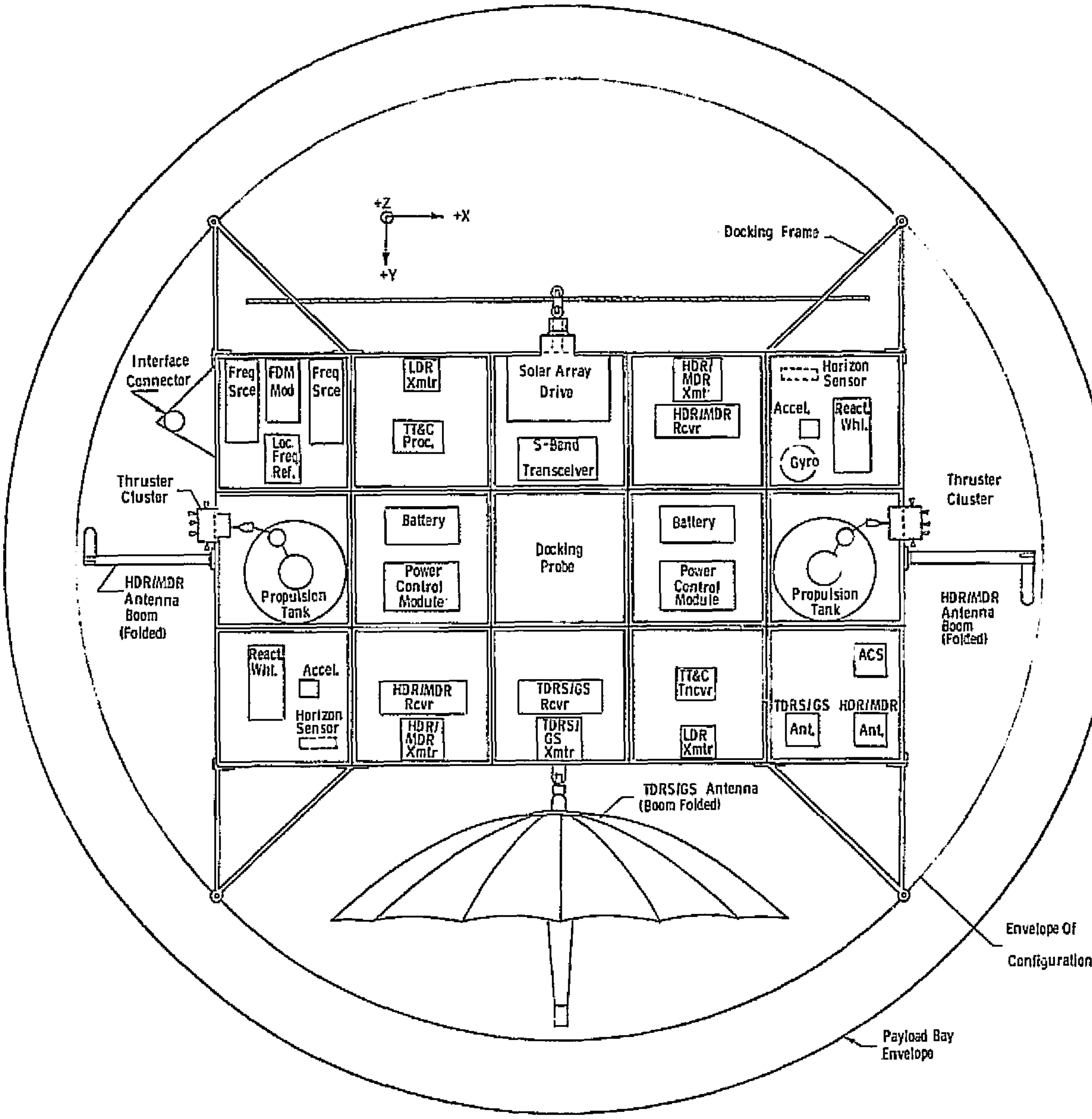

Figure IVC-12 Rear Layout - Serviceable TDRS

Table IVC-8 Module Summary - Serviceable TDRS

			Weight (lbs)	Power (watts)
Module 1 - S-Band Array			30.9	34.0
Frequency source (2)	8.0	16.0		
FDM module	1.3	10.0		
Local frequency reference	13.6	8.0		
Module 2 - LDR			16.8	68.0
Transmitter	7.1	66.0		
TT&C Processor	9.7	2.0		
Module 3 - Solar Array and S-Band Beacon			15.5	5.0
Solar array drive	10.0	3.0		
Transceiver	5.5	2.0		
Module 4 - HDR/MDR			24.0	60.2
Receiver	9.9	8.2		
Transmitter	14.1	52.0		
Module 5 - ACS			23.5	9.0
Horizon sensor	5.5	0.5		
Reaction wheel	10.1	6.0		
Accelerometer	2.2	0.5		
Gyro	5.7	2.0		
Module 6 - RCS			35.7	
Thruster cluster	5.8			
Tank/hardware	10.3			
GN_2	0.2			
Propellant	19.4			
Module 7 - Electrical			39.0	15.0
Battery	22.0			
Power control	17.0	15.0		
Docking Probe Space				
Module 9 - Electrical			33.6	5.0
Battery	22.0			
Power control	11.6	5.0		
Module 10 - RCS			35.7	
Thruster cluster	5.8			
Tank/hardware	10.3			
GN_2	0.2			
Propellant	19.4			
Module 11 - ACS			17.8	7.0
Horizon sensor	5.5	0.5		
Reaction wheel	10.1	6.0		
Accelerometer	2.2	0.5		
Module 12 - HDR/MDR			24.0	60.2
Receiver	9.9	8.2		
Transmitter	14.1	52.0		
Module 13 - TDRS/GS			25.9	45.9
Receiver	4.8	5.3		
Transmitter	21.1	40.6		
Module 14 - LDR			11.1	74.5
Transmitter	7.1	66.0		
TT&C Transceiver	4.0	8.5		
Module 15 - Electronics			20.7	11.0
TDRS/GS antenna	5.1	4.0		
HDR/MDR antenna	10.1	4.0		
ACS module	5.5	3.0		

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Table IVC-9 TDRS Serviceable Version Weight Summary

Item	Weight (lbs)	Item	Weight (lbs)
Module			
1	30.9	Structure	190.0
2	16.8	Module enclosures	280.0
3	15.5	Docking frame	48.0
4	24.0	Interface connector	5.0
5	23.5	Wiring, etc (distributed)	93.0
6	35.7	HDR/MDR antennas (2)	66.4
7	39.0	TDRS/GS antenna	14.4
		S-band array	31.8
9	33.6	S-band beacon	0.2
10	35.7	Omni-antennas (4)	2.6
11	17.8	Solar aspect sensor	3.3
12	24.0	Solar array	38.2
13	25.9	TOTAL	1127.1
14	11.1		
15	20.7		

Table IVC-10 TDRS Replaceable Units

Modules*		
1		50.9 lbs
2		36.8
3		77.2
Solar array drive	10.0	
S-band transceiver and beacon	5.7	
Solar aspect sensor	3.3	
Solar array	38.2	
Encl sure	20.0	
	77.2	
4		44.0
5 (without horizon sensor)		38.0
6		55.7 (36.1 dry)
7		59.0
9		53.6
10		55.7 (36.1 dry)
11 (without horizon sensor)		32.3
12		44.0
13		45.9
14		31.1
15		40.7
HDR/MDR Antenna 1		33.2
HDR/MDR Antenna 2		33.2
TDRS/GS Antenna		14.4
TOTAL		745.7 (706.5 dry)

*Includes 20 lb enclosures.

6) Four of the TT&C omni antennas from the original booster-launched version can be eliminated with Tug boost.

7) Because of the layout of the antennas and the small solar array size, a single solar array was chosen. Differential solar pressure would be slight.

8) The solar array, array drive, S-band beacon/transceiver, and solar aspect sensor would all be replaceable as a single unit.

9) Because the antennas are directional and the drive motors are subject to failure, the antennas should be replaceable. The point of connection should be near the drive motor/antenna vertex.

10) The total satellite weight is 1127 lbs.

11) The maximum weight of replaceable units is 746 lbs.

5. Earth Observations Geosynchronous Platform

a. Configuration

NOTE: Configuration and systems requirements for the Earth Observations Geosynchronous Platform (EOGP) were taken from the Geosynchronous Platform Definition Study, SD73-SA-0036 (Contract NAS9-12909), Space Division, Rockwell International, June 1973, as directed in the Orbital Assembly and Maintenance Study Statement of Work.

The EOGP is one of several geosynchronous platforms derived in the RI studies. These platforms were developed with the idea of gathering multiple functions into a few satellites to reduce satellite traffic. The EOGP combines all earth resources, earth physics, atmospheric, and meteorology objectives into one all-purpose earth observation facility. Table IVC-11 presents a list of the sensor functional acquisitions and the range of observational capabilities.

The geosynchronous platforms were developed with the idea also of using a support ring containing that operational equipment required to maintain the satellite in orbit. Equipment was selected for the support ring that was common to and would support the requirements for all platforms (with minor changes).

Table IVC-11 Earth Observations Sensor Functional Summary

EQUIPMENT ITEM	PRIMARY FUNCTIONS																				
	LITHOLOGY	GEOLOGY	EARTH STRUCTURE	TOPOGRAPHY	WATER RUN-OFF	OCEAN POLLUTION	SNOW/ICE	VEGETATION	OCEAN COLOR/LIFE	SURFACE TEMPERATURE	RESOURCE RECOGNITION	LAND USE	MOISTURE BOUNDARIES	SOIL TEXTURE	SEA STATE	CLOUD CONTENT	AURORA/AIRGLOW	LIGHT POLARIZATION	STORM MAPPING	AIR POLLUTION	MAGNETOSPHERE WAVE/PARTICLES
MULTISPECTRAL TV	X		X	X	X	X		X	X		X	X								X	
MULTISPECTRAL IR SCANNER	X	X	X	X	X	X		X	X	X	X	X	X							X	
PASSIVE MICROWAVE SCANNER		X	X		X	X	X			X	X		X	X							
MULTISPECTRAL RADIOMETER					X	X		X	X	X		X	X	X	X						
AERONOMY SPECTROMETER																	X		X	X	
SPECTRAL POLARIMETER																X	X	X		X	
SFERICS DETECTOR																X	X		X		
ABSORPTION SPECTROMETER																	X			X	
CLOUD CAMERA (3-COLOR SCAN)																X			X		
HIGH-RESOLUTION IR CAMERA						X		X	X	X										X	
ELECTRON/ION MONITOR																					X
LF-VLF TRANSCEIVER																					X
PARTICLE DETECTOR																					X
FLUXGATE MAGNETOMETER																					X
PHOTOMETRIC CLUSTER																					X
INTERFEROMETER/SPECTROMETER																	X				X
SCANNING/GRATING SPECTROMETER																	X				
EUV SPECTROMETER																					X
COSMIC DUST SENSOR																					X
MICROWAVE RADIOMETER	X						X			X	X	X	X	X		X					
SCATTEROMETER RADIOMETER		X	X				X			X		X	X	X	X						
MULTISPECTRAL SPECTROMETER		X	X		X			X			X	X									

Equipment necessary for the particular mission experiment objectives would be placed in separate equipment rings.

Views of the EOGP are presented in Figures IVC-13 and IVC-14.

The major element of the facility is the 1.5-meter Cassegrain-type telescope to provide the capability to meet the resolution required for imaging from geosynchronous altitude. The telescope will provide a footprint less than 100 miles and resolution in the order of 100 feet. The telescope provides a high-resolution capability for various IR and spectral optical, electronic imaging sensors used for mapping, and resource and pollution identification. Three large antennas provide the desired high resolution for microwave scanning, and provide boundary mapping, or surface and atmosphere temperatures, and moisture delineation. Equipment rings are mounted fore and aft on the 1.5-meter telescope barrel. The aft (-Z) equipment ring is at the focal point of the telescope (behind the primary optics) where various optical sensors are coupled to the telescope light beam. Other sensors are mounted in this ring for viewing ambient magnetospheric phenomena. Two forward equipment rings house supporting subsystems (common support module) and direct earth-viewing sensors. These rings also act as a sunshade for the telescope, which looks directly through the core section. Three large dish antennas (30, 30, and 60 feet) and a large (123 feet tip-to-tip) dipole antenna dominate the exterior appearance. In addition, several smaller sensors are externally mounted to provide for separation from the spacecraft or, as in the case of the scan-platform assembly, to enable pointing capability. A docking assembly is provided at both ends (+Z and -Z) to accommodate the servicing system for exchange of replaceable modules.

The particular configuration derived for the platforms (enclosed torroidal rings with internal module installation) was based on the idea that this configuration would accommodate shirtsleeve intravehicular (IVA) maintenance by installing hatches and pressurization capabilities. This capability could be attained after the platform is in orbit by docking hatch assemblies at the existing docking ports.

A breakdown of the equipment and structural weights of the EOGP is presented in Table IVC-12. The total satellite weight is 8,491 lbs. The power requirements are also presented in Table IVC-12. The average operational power requirements are 1,896

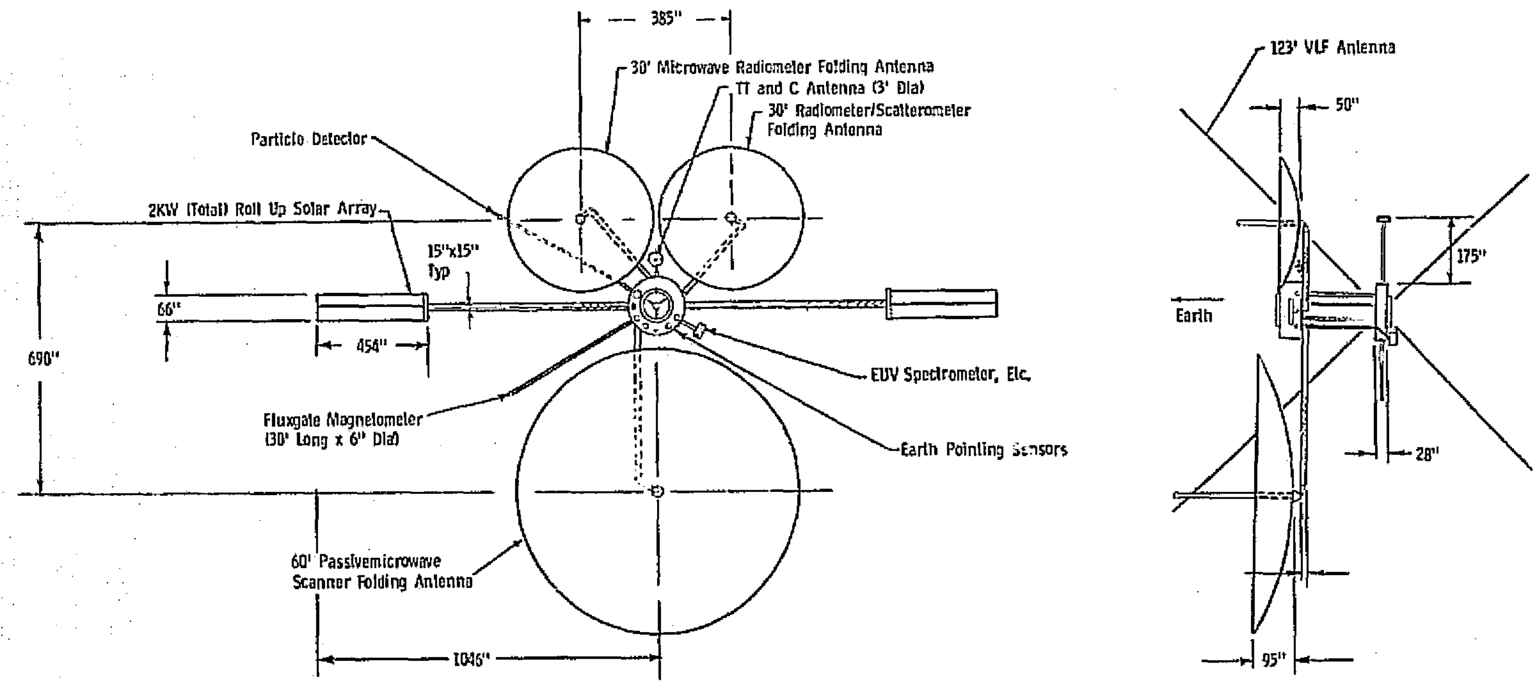


Figure IVC-13 EOGP Overall View

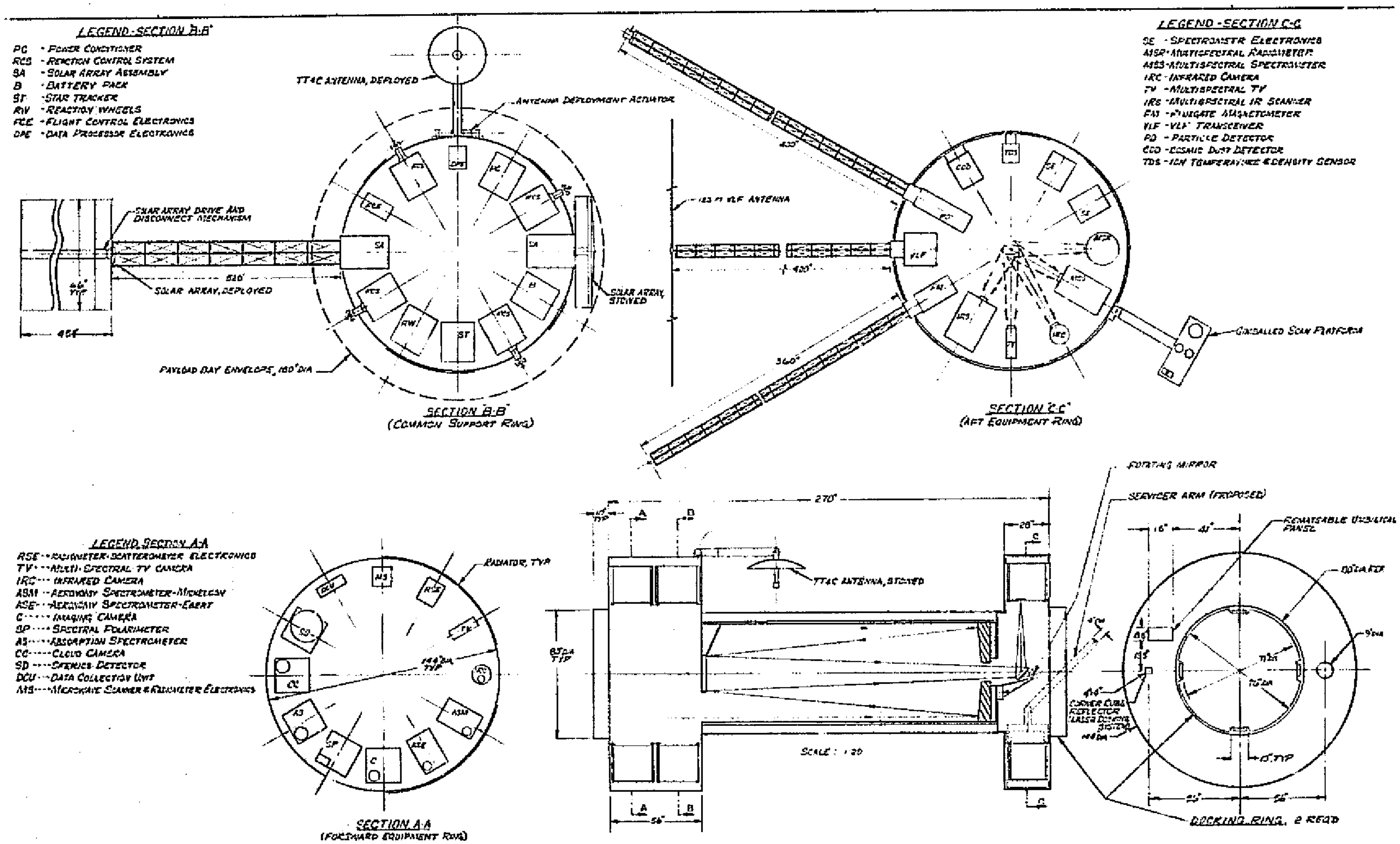


Figure IVC-14 EOGP Details

Table IVC-12 EOGP Weight and Power Summary

Item	Weight (lbs)	Power (watts)
Forward Equipment Ring (Earth Pointing)		
*1. Radiometer-scatterometer electronics	90	150
2. Multi-spectral TV camera	110	200
3. High-resolution IR camera	45	26
Astronomy Spectrometers		
4. Michelson - electronics	43	130
5. Ebert - electronics	58	130
6. Electronic imaging camera	10	21
7. Spectral polarimeter	40	20
8. Absorption spectrometer	30	15
9. Cloud camera	14	15
10. Sferics detector	20	6
11. Data collection unit	11	8
12. Passive microwave scanner electronics	50	175
Microwave radiometer electronics	57	60
Thermal Protection		
Radiator (36 ft² for 524 watts average power), mounts, coldpates, etc.	65	
Structure and Electrical Cabling	1,100	
Docking mechanism	100	
Sferics detector antenna	5	
SUBTOTALS	1,848	974 (524 avg.)
Common Support Ring		
1. Power conditioner	144	20
2. Reaction control system (RCS) (includes 44 lbs propellant)	70	5
3. Solar array assembly (1 kw each)	270	5
Array 42 lbs		
Drive 35		
Storage 35		
Shunt dissipator 8		
Boom (astromast) 150		
4. Battery pack (8)	392	40
5. RCS	70	5
6. Star trackers	30	20
7. Reaction wheels	75	30
8. RCS	70	5
9. Solar array assembly	270	5
10. Flight control electronics	44	50
11. RCS	70	5
12. Data processor electronics	60	28
Common Support Ring (Continued)		
Thermal Protection	39	
Radiator (22 ft²) 21		
Mounts and supports 5		
Coldplates, heat pipes, and interface material 13		
Structure	400	
Primary 100		
Secondary 300		
Electrical Cabling (distributed)	175	
Data Processor TT&C Antenna	10	
SUBTOTALS	2,169	218
Telescope Core		
1.5-meter telescope (includes rotating mirror)	1,000	75
Microwave radiometer (30' diameter)	70	
Extension boom	80	
Radiometer-scatterometer (30' diameter)	70	
Extension boom	80	
Passive microwave scanner (60' diameter)	200	
Extension boom	110	
SUBTOTALS	1,610	75
Aft Equipment Ring		
1. Spectrometer electronics	75	465
Scanning-grating 68		
EUV 7		
2. Spectrometer electronics	75	465
Interferometer 68		
EUV 7		
3. Multispectral radiometer	40	20
4. Multispectral spectrometer	175	170
5. High resolution IR camera	45	24
6. Multispectral TV	110	200
7. Multispectral IR scanner and support electronics	200	130
8. Fluxgate magnetometer	3	20
Extension boom	55	
9. LF-VLF transceiver	27	70
10. Particle detector	75	50
Extension boom	55	
Aft Equipment Ring (Continued)		
11. Cosmic dust detector and support electronics	21	10
12. Ion temperature and density sensor	13	12
LF-VLF dipole antenna	60	
Extension boom	70	
Thermal protection	130	
Radiator (72 ft² for 1,054 watts avg. power), mounts, coldplates, etc.		
Structure and electrical cabling	1,100	
Docking mechanism	100	
SUBTOTALS	2,429	1,636 (1,054 avg.)
Scanning Platform		
EUV Spectrometer	25	
Scanning grating spectrometer	186	
Interferometer spectrometer	166	
Photometric cluster	38	25
SUBTOTALS	415	25
Summary		
Forward Equipment Ring	1,848	974 (524 avg.)
Common Support Ring	2,169	218
Telescope Core	1,610	75
Aft Equipment Ring	2,429	1,636 (1,054 avg.)
Scanning Platform	415	25
TOTALS	8,491	2,928 (1,696 avg.)

*Numbers indicate clock locations of modules (viewed from the earth-pointing end).

watts. Solar arrays sized for 2,000 watts are used to generate electrical power. Heat is dissipated by the use of solid heat pipes and variable conductance heat pipes. These carry the heat to radiators mounted on the external surfaces of the equipment rings.

Table IVC-13 summarizes the weights and sizes of the replaceable units. The total weight of all replaceable units is 4,287 lbs. The standard size of the internal modules is 24 x 20 x 24 inches. Modules, however, may vary in size. Some of the modules require a length up to 36 inches. Figures IVC-15 and IVC-16 present views of proposed modules and the standard module attachment mechanism.

Special provisions for the rotating mirror assembly are needed to permit extraction of modules in the aft equipment ring. Standard module attachment mechanisms are proposed to enable removal and temporary storage of this assembly prior to maintenance in this area (see Figure IVC-14). After maintenance, the mirror assembly (or a new spare) would be replaced. Guide pins in the base would be used to assure precise alignment.

As seen in Figure IVC-14 for the present design, there is, in some cases, inadequate clearances between modules during extraction. Modules would have to be designed to assure adequate clearances.

To accommodate shirtsleeve replacement of internal modules, the RI study proposed schemes such as that presented in Figure IVC-17 to seal openings in the equipment rings prior to pressurizing the satellite interior. The example presented showed a method for motor actuating or manually cranking a cover to close the RCS thruster port. On the EOGP there will be 9 places that equipment extends through the pressure wall. In all cases except the solar array booms, the RI method for sealing the opening might be feasible. In the case of the solar array assembly, this is not possible. The solar array and storage canister would have to be removed externally. Further effort is needed to develop the design requirements for the EOGP to permit pressurizing the interior for shirtsleeve maintenance.

Investigations of maintenance methods and requirements for shirtsleeve operations would be very similar to those for EVA pressurized-suit operations. In either case, pressurized-suit EVA would be required for maintenance outside of the satellite. In either case, hand tools would be required to assist in unlatching

Table IVC-13 EOGP Replaceable Units

Item	Weight (lbs)	Packaged Dimensions (inches)
Internal Replacement		
Forward Equipment Ring Modules:		
1. Radiometer-scatterometer electronics	90	12 x 12 x 15
2. Multispectral TV camera	110	6 x 18 x 24
3. IR camera	45	12 dia x 24
4. Aeronomy spectrometer-Michelson	43	24 x 16 x 24
5. Aeronomy spectrometer-Ebert	58	24 x 16 x 24
6. Imaging camera	10	6 x 6 x 14
7. Spectral polarimeter	40	8 x 20 x 24
8. Absorption spectrometer	30	20 x 20 x 24
9. Cloud camera	14	9 x 9 x 10
10. Sferics detector	20	19 x 17 x 11
11. Data collection unit	11	18 x 18 x 6
12. Microwave scanner and radiometer electronics	107	12 x 12 x 24
Common Support Ring Modules:		
1. Power conditioner	144	24 x 20 x 24
2. Reaction control system (RCS)	70	24 x 20 x 24
3. Solar Array Assembly* (SA)	270	24 x 20 x 24 int. 10 dia x 66 ext.
4. Battery pack	392	24 x 20 x 24
5. RCS	70	24 x 20 x 24
6. Star trackers	30	24 x 20 x 24
7. Reaction wheels	75	24 x 20 x 24
8. RCS	70	24 x 20 x 24
9. SA	270	24 x 20 x 24 int. 10 dia x 66 ext.
10. Flight control electronics	44	10 x 22 x 18
11. RCS	70	24 x 20 x 24
12. Data processor electronics	60	11 x 14 x 20
Aft Equipment Ring Modules:		
1. Spectrometer electronics	75	24 x 20 x 16
2. Spectrometer electronics	75	24 x 20 x 16
3. Multispectral radiometer	40	20 dia x 24
4. Multispectral spectrometer	175	20 x 18 x 30
5. IR camera	45	12 dia x 24
6. Multispectral TV	110	6 x 18 x 24
7. Multispectral IR scanner	200	24 x 20 x 30
8. Fluxgate magnetometer	58	8 x 8 x 24
9. VLF transceiver	27	20 x 20 x 24
10. Particle detector	130	12 x 12 x 36
11. Cosmic dust detector	21	20 x 15 x 19
12. Ion temp. and density sensor	13	12 x 6 dia
Above 6 Rotating Mirror Assembly	10	TBD
External Replacement		
Data Processor TT&C Antenna	10	10 x 36 dia.
Microwave Radiometer Antenna	150	120 x 30 dia.
Radiometer-Scatterometer Antenna	150	120 x 30 dia.
Microwave Scanner Antenna	310	120 x 30 dia.
LF-VLF Dipole Antenna	130	3 x 3 x 3
Scanning Platform	415	36 x 46 x 69
TOTAL	4,287	

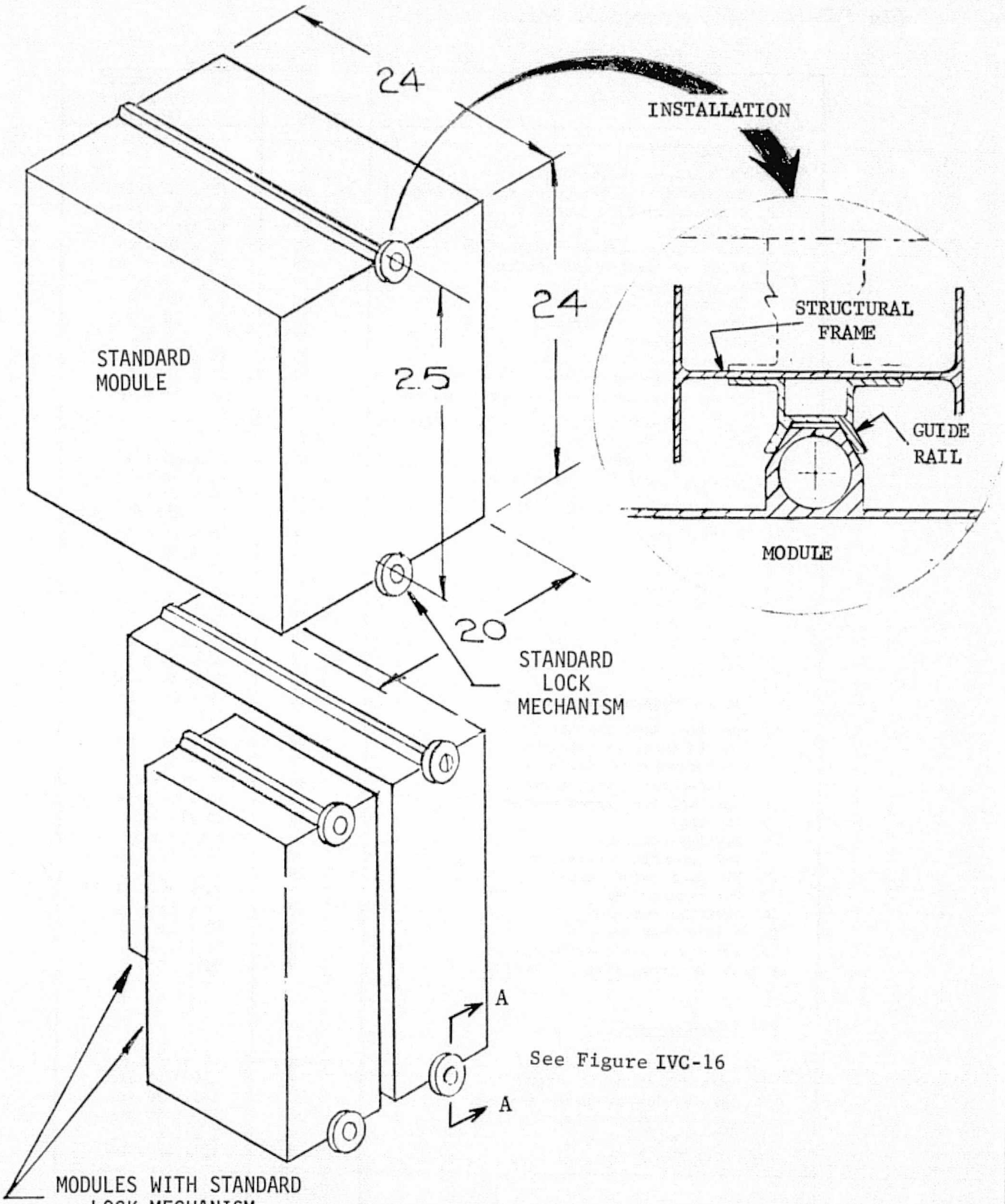
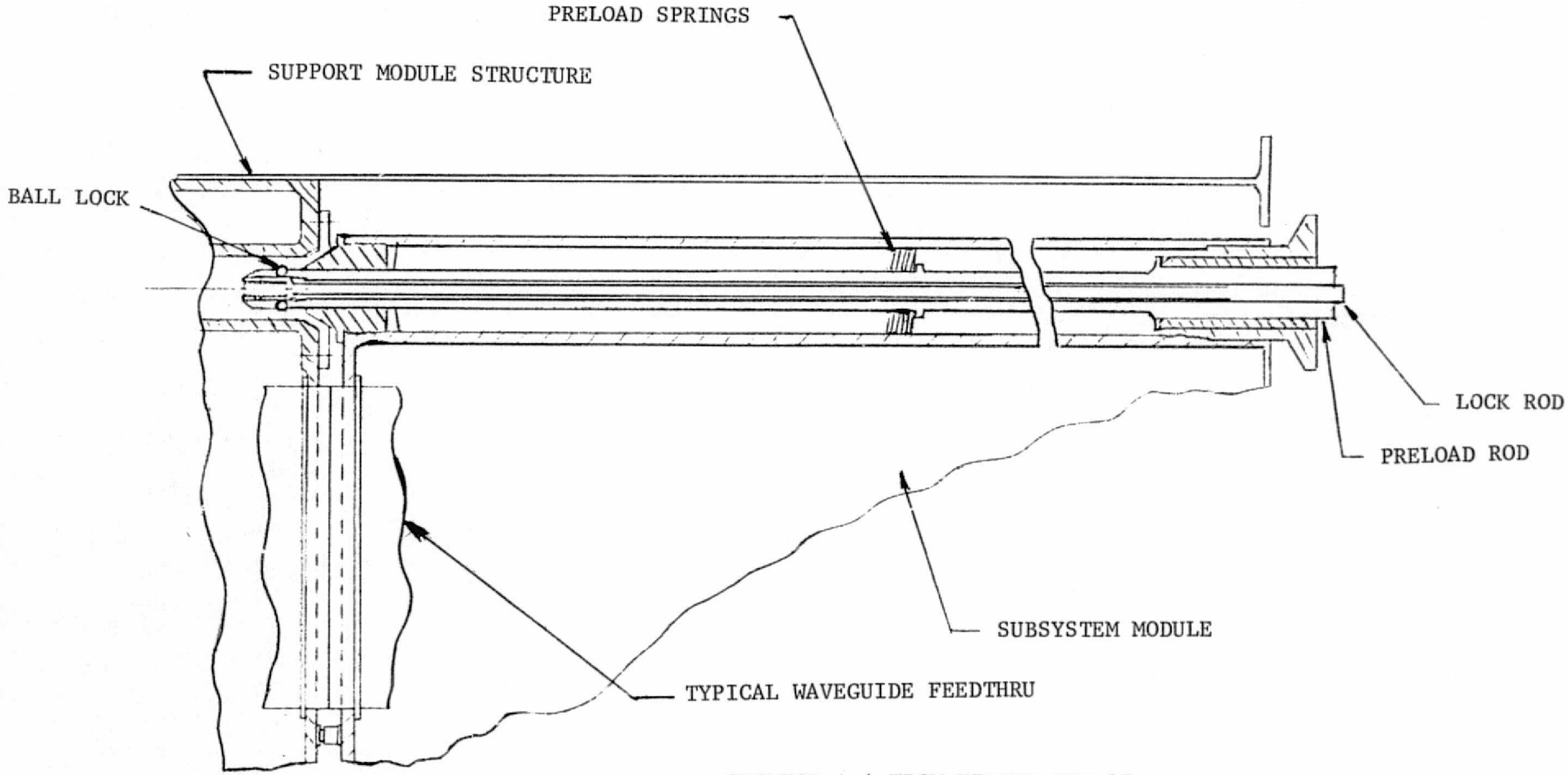


Figure IVC-15 EOGP Standard Modules



SECTION A-A FROM FIGURE IVC-15

Figure IVC-16   EOGP Module Attachment Mechanism

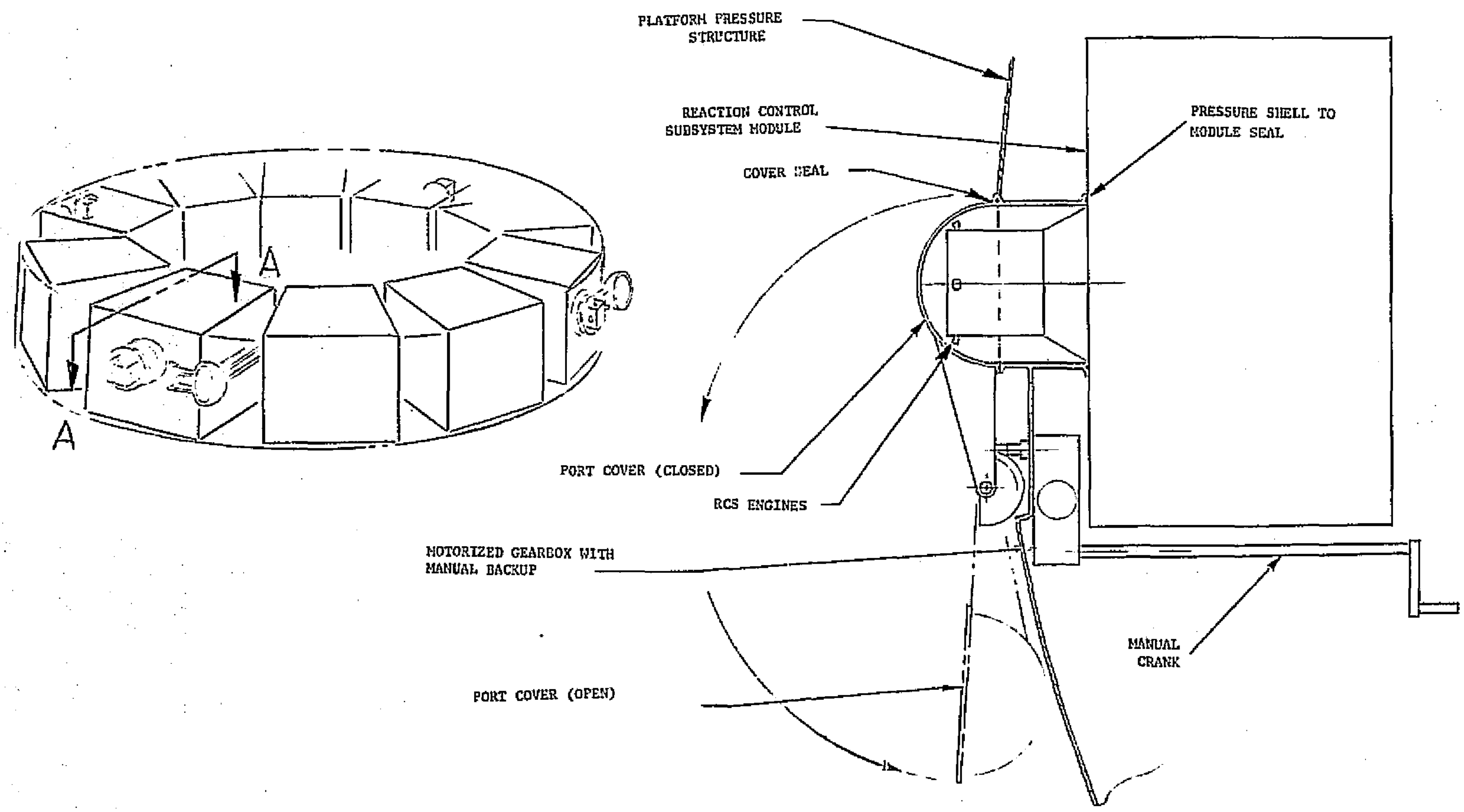


Figure IVC-17 EOGP Shirtsleeve Serviceable RCS Concept

the module attachment mechanisms. Manipulating and transporting the modules would be somewhat easier in shirtsleeve operations. Additional quantities of oxygen would be required for pressurizing the satellite for IVA operations, however a comparable quantity of oxygen would be required for EVA operations. Optical surfaces could be contaminated by body products in either case; however, it would be easier to control this by EVA operations by ducting the pressure suit effluent away from the maintenance area. The IVA shirtsleeve maintenance approach will not be investigated in detail in this study, however pertinent IVA comparisons will be discussed, where applicable, in the EVA maintenance approach discussed later.

6. Radio Astronomy Telescope

Replaceable subsystem modules on the radio astronomy telescope will be located as presented in Figure IVC-18. Two star trackers (one with TT&C antenna) and four ACS propellant pods will be located at "Y" joints on the antenna rib structure. Remaining replaceable subsystem modules will be located in the end of the central core. Three solar array assemblies are assumed. Because of the small solar power requirement, the solar arrays will be small and easily shaded by the central core. Use of three small arrays will permit adequate solar viewing by the cells of one or more of the arrays at all times regardless of the various pointing directions of the telescope. If the telescope net support material is opaque to solar radiation, then the solar arrays at the bottom of the core could be in shadow much of the time. In this case, additional arrays would be needed on the mast above the net. Changeout of these would be similar to that for the ACS modules. Weight summaries are presented in Table IVC-14.

7. Satellite Conceptual Designs Summary

A summary of the weight and size characteristics of the satellite replaceable units is presented in Table IVC-15.

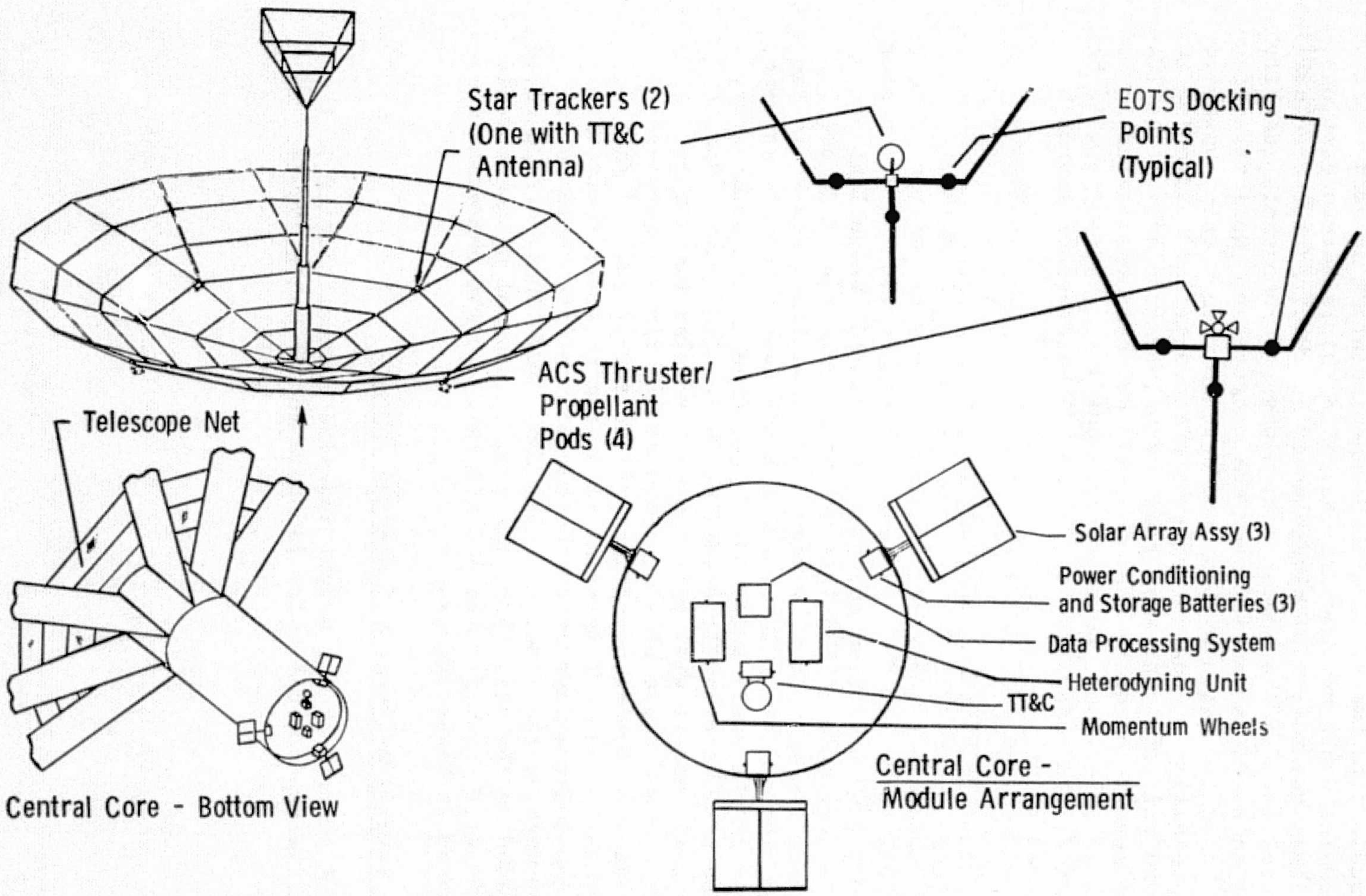


Figure IVC-18 RAT Replaceable Subsystems

Table IVC-14 Radio Astronomy Telescope Replaceable Modules Weights

Module	Quantity	Unit Weight (lbs)	Total Weight (lbs)
Internal - Core			
Power Generation Solar array (75 watts) Solar array drive Sun sensors Power conditioning Storage battery	3	50	150
Data Processing System	1	60	60
Heterodyning Unit	1	120	120
TT&C System (with directional antenna)	1	90	90
Momentum wheels and electronics	1	120	120
External			
ACS Propellant and Thruster Pod	4	60	240
Star Tracker	1	15	15
Star Tracker with TT&C Antenna	1	35	35
TOTALS	13		830 lbs

Table IVC-15 Summary of Replaceable Units

Satellite	Baseline Satellite Weight (lb)	Satellite Weight (lb)	Weight of Total Spares Complement (lb)	Volume of Total Spares (Assuming 0.05 ft^3/lb) (ft^3)	Maximum Number of Modules	Largest Module Weight (lb)	Largest Module Envelope Dimensions (ft)
DWS	1,284	1,904	1,329	66.5	14	222 Solar Array Assembly	2x2x8.5
TDRS	669	1,139	758	37.9	17	77 Solar Array Assembly	2x6x9
Intelsat	3,245	2,710	1,986	99.3	22	135 Solar Array & Propulsion Module	2x4x7 2x2x2
SEOS	3375 - per SSPD 3760 - per PUT Study	3,697	1,782	89.1	12	206 Attitude Control System	3.5x3x1.7 Truncated Pyramid x2 Thick
EOGP	NA	8,491	4,287	214.4	43	392 Battery Pack 415 External Scanner 310 Antenna	2x2x2 3x4x6 2.5 Dia x 10
Radio Astronomy Telescope	NA	27,000	830	41.5	13	120 Momentum Wheels & Electronics	1x2x2

D. PROCEDURES AND TECHNIQUES (TASK 5)

1. Maintenance Approaches

The following three maintenance mission approaches were analyzed for the subject satellites:

Approach 1 - Maintenance in Geosynchronous Orbit using Reusable Tug/Servicer

Approach 2 - Maintenance in Geosynchronous Orbit via EVA from Manned Servicing Module (MSM)

Approach 3 - Maintenance in Shuttle Orbit using Shuttle Remote Manipulator System (RMS) and EVA

Approach 1 requires only one Shuttle/Tug flight to place a servicer in geosynchronous orbit and return it. The following operational steps are depicted in Figure IVD-1.

Approach 1 Operational Steps:

1) Tug transfers servicer to satellite orbit and docks.
2) Servicer connects umbilical and deactivates satellite.
3) Servicer performs maintenance activities by preprogrammed direction or man-remote ground control.
4) Tug orients assembly to ground pointing.
5) Servicer activates satellite.
6) Preliminary checks performed by ground controllers.
7) Tug/Servicer separates from satellite.
8) Final satellite functional checks.
9) Tug/servicer returns to Shuttle Orbiter.

The concept of using an Earth Orbital Teleoperator System (EOTS) as the servicer in Approach 1 will be discussed in Section IV.D.10.

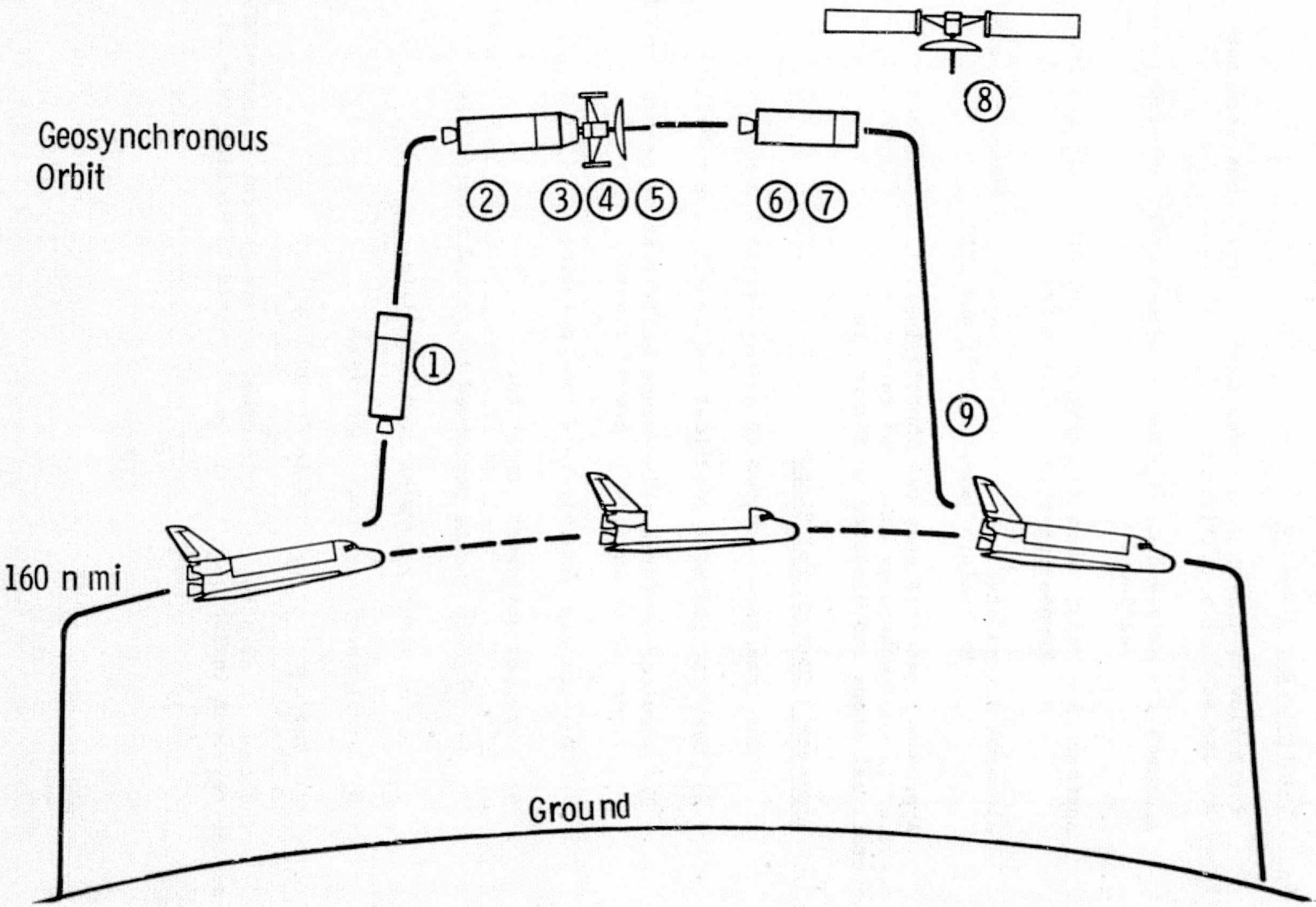


Figure IVD-1 Maintenance Mission - Approach 1

Approach 2 requires two Tugs in tandem. The first Tug will place the total assembly in an elliptical phasing orbit of about 160 x 7000 n mi. During the first orbit, the Tugs will separate. At perigee, the first Tug will burn to return to the Orbiter. The second Tug will burn into a transfer orbit to geosynchronous altitude. The following operational steps are depicted in Figure IVD-2. Figure IVD-3 presents a view of an MSM concept.

Approach 2 Operational Steps:

1) Tandem Tugs transfer manned servicing module (MSM) to satellite orbit and docks.

2) Crewmen connect umbilical and deactivate satellite.

3) EVA crewmen perform maintenance activities.

4) Tug orients assembly to ground pointing.

5) Crewmen activate satellite.

6) Preliminary satellite checks performed by ground controllers.

7) Tug/MSM separate from satellite.

8) Final satellite functional checks.

9) Tug/MSM return to Shuttle Orbiter.

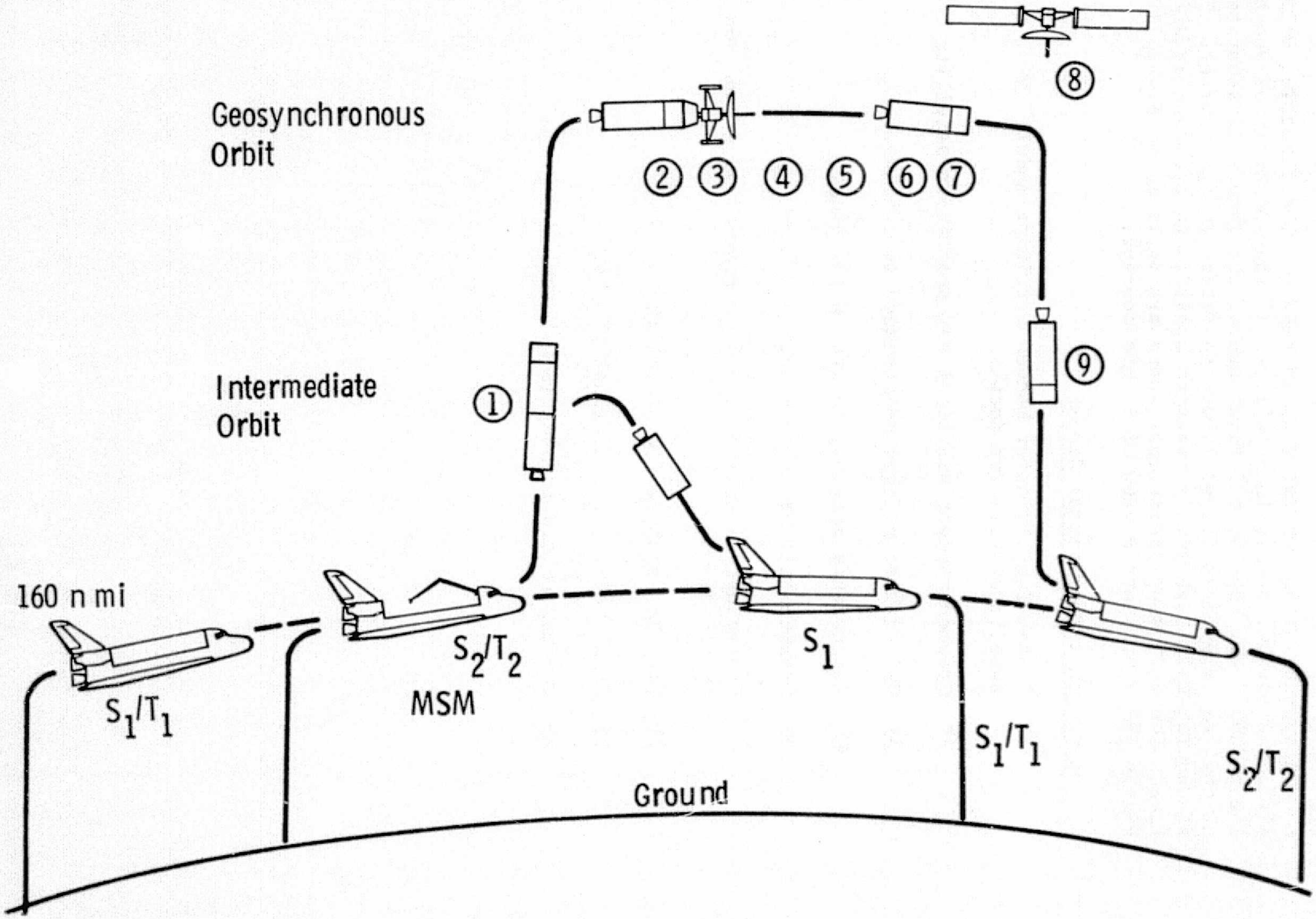
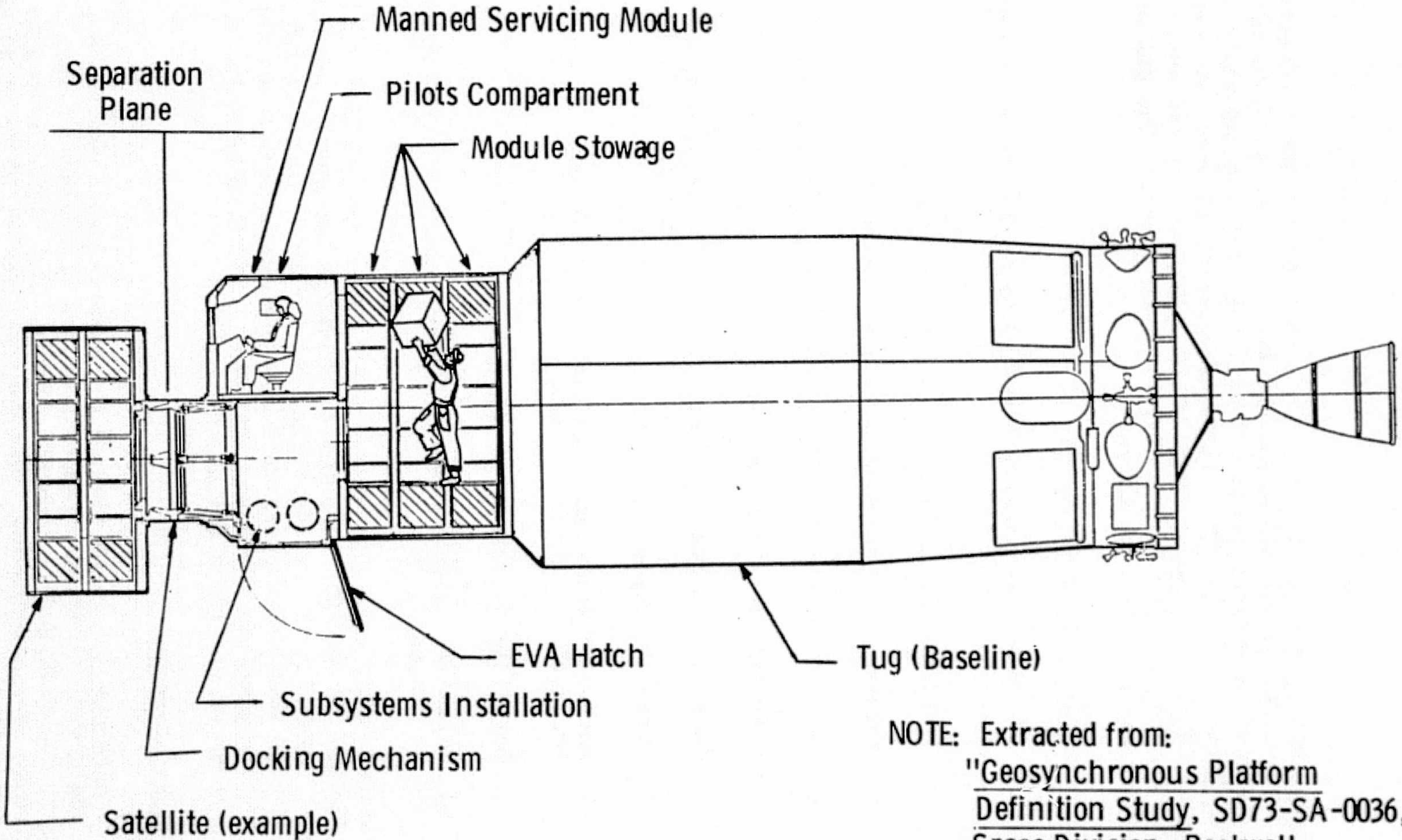


Figure IVD-2 Maintenance Mission - Approach 2



NOTE: Extracted from:
"Geosynchronous Platform Definition Study, SD73-SA-0036, Space Division, Rockwell International, June 1973

Figure IVD-3 Manned Servicing Module

Approach 3 requires two Tugs; one to retrieve the satellite from geosynchronous orbit and another to deploy the serviced satellite. Larger satellites require 3 or 4 Tugs to retrieve and return the satellite to orbit. The following operational steps are depicted in Figure IVD-4. For maintenance at the orbiter, the RMS would be used as a work platform (Figure IVD-5) or to transfer the EVA crewmen and/or spares to the worksite.

Approach 3 Operational Steps:

1) Tug retrieves satellite from Geosynchronous orbit (appendages folded).

2) Satellite/Tug docked to orbiter using RMS.

3) Maintenance performed using RMS and/or EVA.

4) Orbiter orients satellite to ground pointing.

5) Satellite activated and appendages deployed.

6) Satellite checks performed by ground controllers.

7) Satellite appendages folded.

8) Satellite transferred to loaded Tug delivered by second orbiter.

9) Satellite delivered to geosynchronous orbit.

10) Satellite systems checks performed prior to Tug return to orbiter.

The following sections discuss the mission budgets and requirements for maintenance of each of the subject satellites. Baseline Tug boost capabilities are investigated in terms of the mass of spare modules that can be carried to geosynchronous orbit and returned. More specific missions are discussed later, wherein spares replacement is based on predicted failure rates.

2. DWS Maintenance

a. Approach 1 - It is assumed the maintenance tasks to be performed are known. A servicer capable of the maintenance tasks (modular changeout, non-modular replacement, repair, etc.) is delivered to the DWS orbit by the baseline Tug. The Tug/servicer assembly is docked to the DWS. The maintenance tasks are performed. The Tug/servicer is returned to the Shuttle orbiter and subsequently to the ground.

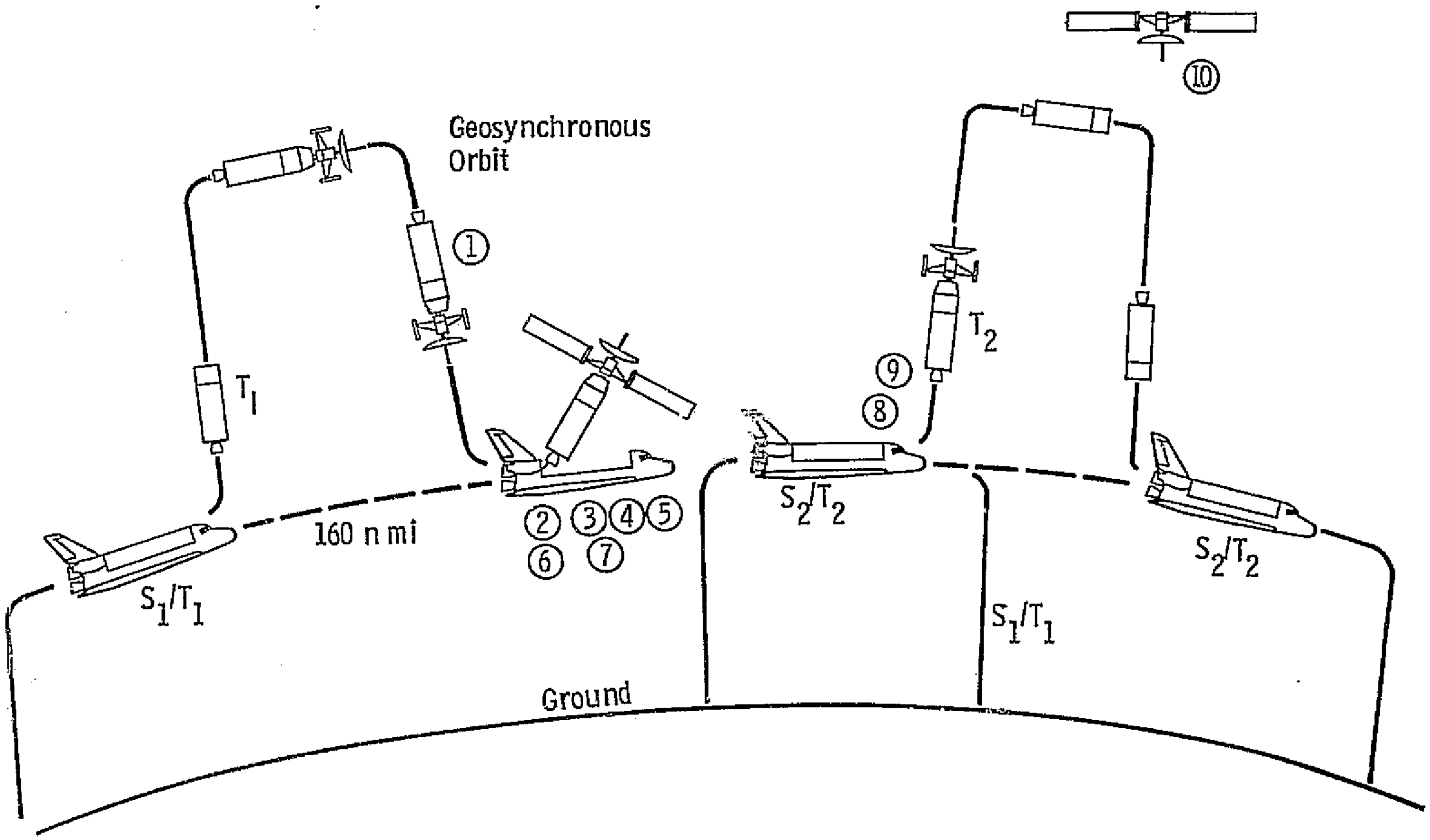


Figure IVD-4 Maintenance Mission - Approach 3

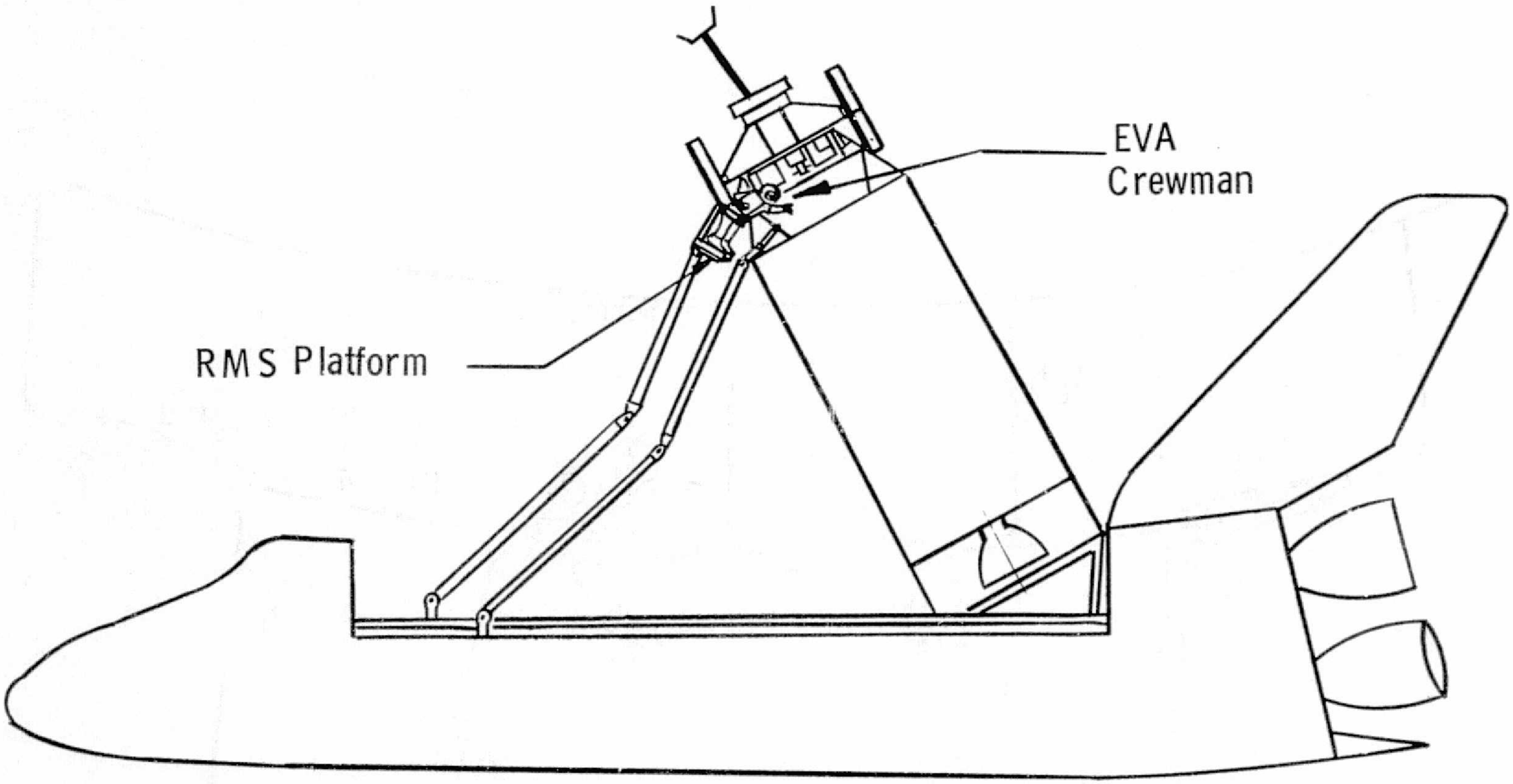


Figure IVD-5 Satellite Maintenance Using RMS and EVA

The general mission sequence and primary operations would be:

1) Tug/servicer transfer to DWS orbit.

2) Tug automatic rendezvous/docking maneuvers initiated.
   - Solar arrays folded on DWS (ground command)
   - Inhibit DWS RCS
   - Docking completed
   - Verify secure

3) Umbilical lockup (power, comma d, data, etc.).

4) Deactivate DWS systems.

5) Wait until residual reaction wheel momentum is removed (2 hours). Tug maintain attitude control.

6) Perform nonmodular replacements or repairs using manipulators/TV system under ground command.

7) Perform modular replacement using preprogrammed automatic servicer, initiated by ground command.

8) Reorient assembly to functional attitude.

9) Deploy solar arrays. Power up DWS (except RCS).

10) Perform preliminary checkout.

11) When reaction wheel speed is up, undock Tug/servicer separate to safe distance and loiter.

12) Activate DWS RCS.

13) Ground verify proper DWS functional operation.

14) Tug/servicer return to Shuttle orbiter.

Mission Details

Tug/Servicer Transfer to DWS Orbit--The Tug/Servicer will be tilted from the Shuttle payload bay. Using Shuttle power, a computer checkout routine will verify operation of the servicer. The Tug/Servicer will then be deployed using the RMS. On separation to a safe distance, the Tug fuel cells will be started, all Tug and servicer systems activated, and functional operations verified.

At the proper place in the orbit, the first main propulsion system (MPS) boost burn will be initiated to place the Tug/Servicer in an intermediate phasing orbit. After about one revolution, the MPS will be fired to inject the Tug/Servicer into a transfer orbit with the apogee at the geosynchronous altitude. At geosynchronous altitude the MPS is again burned to circularize the orbit of the Tug/Servicer.

Proper selection of the burn points in the phasing orbits will place the Tug/Servicer near the DWS longitudinal location. The 28.5 deg plane change is accomplished by out-of-plane-vector pointing, distributed at each burn. Most of the plane change is accomplished at the geosynchronous circularization burn.

Table IVD-1 presents the mission budgets. The calculations were based on ΔV budgets presented in the MSFC baseline tug documents. A weight of 1150 lbs was assumed for the servicer and 1329 lbs for the maximum weight of spares to be installed. Table IVD-2 presents the weight breakdown of the servicer.

Table IVD-2 Servicer Weight Summary

Docking Mechanism	100
Manipulator and TV Arm	400
Structure	400
Adapter	100
Subsystems	150
	1,150 lbs

Taken from Rockwell Geosynchronous Platform Definition Study SD73-SA-0036, June 1973.

Tug/Servicer Rendezvous and Docking with the DWS--The rendezvous and docking of the Tug/Servicer with the DWS will be accomplished automatically for the most part. On acquisition of the DWS by laser radar, the Tug/Servicer will initiate closure maneuvers. During the last phases of this period, the ground controllers will initiate commands to roll in the DWS solar arrays. Although the booms and arrays should be designed to withstand docking loads, the precaution of rolling the arrays should be taken to preclude excessive bending stresses on the array booms in the event of inadvertent high docking rates. Prior to docking, the Tug/Servicer will hold close to the DWS while the ground controllers verify the predocking configuration by the use of TV. The DWS RCS will be inhibited at this time to preclude any desaturation maneuvers at the time of docking. After verification of proper DWS configuration, the ground controllers will authorize the final closure. On docking, signals from the Tug/Servicer, via telemetry to the ground, will verify capture-latch closures.

Table IVD-1 Approach 1 - DWS Geosynchronous Mission Budgets

Event	Duration (Hours)	Inerts/Losses* (Lbs)	APS (Lbs)	MPS ΔV (Ft/Sec)	Initial Vehicle Weight (Lbs)
Shuttle Ascent and Tug Preseparation	8.0	24.0			58937
Tug Separation from Orbiter	2.0	10.0	8.6		58913
Phase in Shuttle Orbit (160 n.mi.)	11.0	46.0	21.4		58894
Phasing/Plane Change Burn	0.13			4494	58827
Coast 1 Rev. in Phasing Orbit	3.0	5.0	17.4		43189
Inject into Geosynchronous Transfer (Perigee Burn)	0.11			3672	43166
Coast to Midcourse Correction	1.5	6.0	13.8		33534
Midcourse Correction	0.03			50	33514
Coast to 19,300 n.mi. Apogee	3.96	16.0	14.0		33377
Circularize at Geosynchronous Altitude (Apogee Burn)	0.12			5826	33347
Rendezvous and Docking	4.0	15.0	96.5		22339
On-orbit Maintenance and Checkout	13.0	52.0	8.0		22227
Phase at Geosynch for Nodal Crossing	11.4	45.0	11.2		22167
Deboost Burn	0.08			5840	22111
Coast to Midcourse Correction	1.0	6.0	7.5		14798
Midcourse Correction	0.01			13	14784
Coast to 170 n.mi. Perigee	4.2	24.0	8.1		14768
Inject into Return Phasing	0.05			3791	14736
Coast 1 Rev. in Phasing Orbit	3.0	18.0	7.8		11354
Circularize at 170 x 170 n.mi. Orbit	0.05			4243	11328
Rendezvous and Dock with Orbiter	4.0		32.4		8461
TOTALS	62.6	243.0	246.8		

*Boil off, fuel cell consumables

Propellant Consumed	49993
Unused Propellant Capacity	196
APS Consumed	247
Unused APS Capacity	70

Returned Weight		8429
Tug dry weight	5150	
Unusable residuals	576	
APS reserve	29	
Propellant reserve	195	
Servicer	1150	
Used spares	1329	
	8429	

Prepare DWS for Maintenance--On docking, the servicer will automatically engage an umbilical connector and deactivate the DWS systems. The reaction wheel momentum will be removed in about two hours by use of motor braking.

During this and subsequent periods, the Tug will maintain attitude control of the Tug/Servicer DWS assembly. The attitude will be such that adequate Tug communications pointing is maintained as well as DWS orientation for thermal control to prevent excessively low temperatures.

Perform Nonmodular Replacements or Repairs--These maintenance tasks would be performed first since failure of accomplishing these tasks would be more likely than for modular replacements. If the maintenance tasks were unsuccessful, at least the new spares could easily be returned. Nonmodular replacements or repairs would only be planned, however, if there was a high probability of success.

These maintenance tasks would require the use of servicer functions not readily adaptable to automatic operations. The most probable method would be manipulator arms, aided by television, and under ground control. Because of transmission lags, even though small, the use of manipulators in this situation would be a slower process.

Typical tasks in this category are:

- Replacement of the solar array past some point toward the end of the array boom. For the DWS, each solar array replaceable unit includes the liquid slip rings, the drive motor, the cesium ion tip thrusters, and the sun sensors (see Figure IVD-6).

- Installation of a new furled antenna would be possible, but because of the passive nature of the antenna, would not be a likely maintenance task.

- Reshape small structural members, antennas, etc., that are bent during deployment and have caused a degradation in system performance in such a manner that the cause could be identified from data received at the ground.

A period of one hour per exchange of each solar array unit is assumed in the functional timelines.

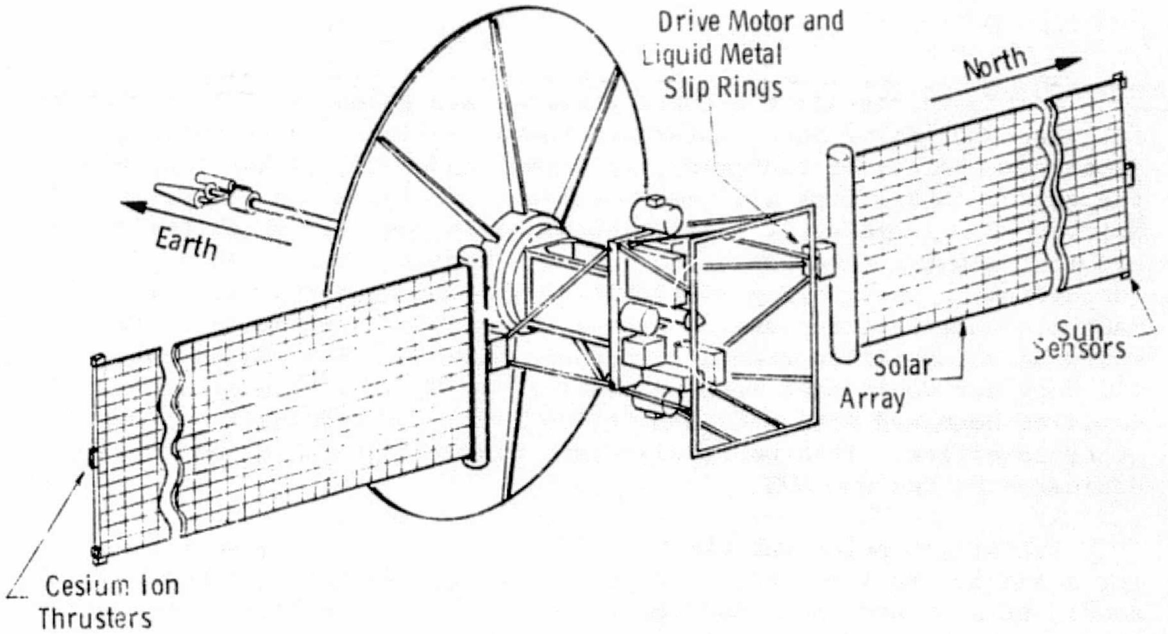


Figure IVD-6 Disaster Warning Satellite-Serviceable

Perform Modular Replacements--The replacement of modules at known locations and close proximity to the servicer would be performed through preprogrammed commands, after an "initiate" command from the ground. The Tug avionics computer could be modified to be the controller on servicer maintenance functions. The specific replacement procedures would be preprogrammed prior to flight for modules known to need replacement. Any subsequent replacements that may be required after checkout (and if the spares were onboard) could be programmed from ground commands.

It would appear feasible that the ground controllers should monitor the automatic procedures via TV to assure proper functioning. Thermal distortions could cause misalignments such that the servicer might not operate properly. The ground controller could intervene and perform the operations by remote control at the expense of time.

A time of 15 minutes per module replacement (12 DWS modules) is assumed for the automatic mode, and 30 minutes per module for the manned remote control mode.

Contingencies--In the event problems are encountered in successfully accomplishing the maintenance tasks, malfunctions occur that jeopardize safety of equipment, or normal undocking is not possible, the ground controllers will make real-time decisions on corrective action. For example, if some maintenance is required which the docked servicer might not be capable of but a subsequent one could perform (depending on Shuttle/Tug availabilities), the decision might be made to continue the planned tasks. The remaining maintenance tasks would be accomplished with a subsequent mission. The subsequent Tug/Servicer would need less spares for the DWS, but the mission could be combined with maintenance, delivery, or retrieval of some other satellite. This would eliminate an additional mission totally dedicated to the one DWS.

The single point failure of a capture-latch failing to release (or a similar failure) would prevent recovery of the Tug. Means should be provided to mechanically (and/or by pyrotechnics) separate the Tug at the servicer interface.

Reactivate DWS Systems--On completion of all maintenance tasks, the solar arrays would be deployed and all DWS systems, except the RCS, would be reactivated. The assembly would be reoriented to the normal DWS pointing attitude. Preliminary checkouts would be performed to verify DWS mission performance. Since the disaster warning network is separate from the NASA network, the National Oceanic Atmosphere Agency (NOAA) personnel would perform the operational checks in conference with the NASA Tug/SC controllers.

Separate and Perform Final Checks--When the reaction wheel speed is up, the Tug/Servicer will undock and separate to a safe distance and the DWS RCS will be activated. The Tug will loiter in close proximity until all systems checks are verified by NOAA and NASA controllers.

Tug/Servicer Return to Orbiter--The deboost burn and return to the orbiter will be accomplished per the budget and schedule presented in Table IVD-1. Final rendezvous and docking using the RMS will be accomplished by the orbiter crew.

Discussion

As seen in Table IVD-1, the DWS maintenance mission, using a 1,150 lb servicer and carrying 1,329 lbs of spares, taxes the capability of the baseline Tug. The propellant reserves were 195 lbs. The desired propellant reserve is 300 lbs. Of course, it is unlikely that a total changeout of DWS replaceable units would be needed, as assumed here.

The mission timeline is presented in Table IVD-3. Because of propellant limitations, no time reserve is available. Any time delays would result in more cryogenic propellant boiloff and a possibility of not completing the total maintenance functions. It is noted that from about hour 10 to hour 66 (56 hours) in the mission, the Shuttle orbiter will be free to be used in other mission operations.

Table IVD-4 details the communications paths during the mission. During LEO operations, the TDRSS will be used. The STDN will be used during geosynchronous orbit operations. No communications incompatibilities are noted.

Table IVD-5 presents the electrical power sources. The Tug has a capability to supply 600 watts to a payload. The power required by the servicer or the DWS during servicer-controlled operations shall be limited to 600 watts.

Table IVD-6 summarizes the various commands during the mission. The capability is needed to control subsystem functions from the ground via telemetry and from the servicer via an umbilical connection.

The following capabilities are required because of this approach to maintenance of the DWS:

DWS

1) Capability to roll in or deploy the solar arrays.
2) Receptacle for umbilical attachment from the servicer.
3) Laser radar reflectors (corner cubes).
4) Replaceable subsystem modules, including solar arrays.
5) Docking frame and latches compatible with the servicer.
6) Solar array booms capable of withstanding docking loads.

Servicer

1) Docking frame compatible with DWS.
2) Servicing system controlled by commands from Tug.
3) Servicing system capable of preprogrammed changeout of DWS modules and remote-control changeout of replaceable units. The latter will incorporate the use of TV.
4) Umbilical system capable of being connected to the DWS to convey control commands and electrical power.
5) Backup means of separation in the event capture latches fail to open.

Table IVD-3 Approach 1 - DWS Mission Timeline

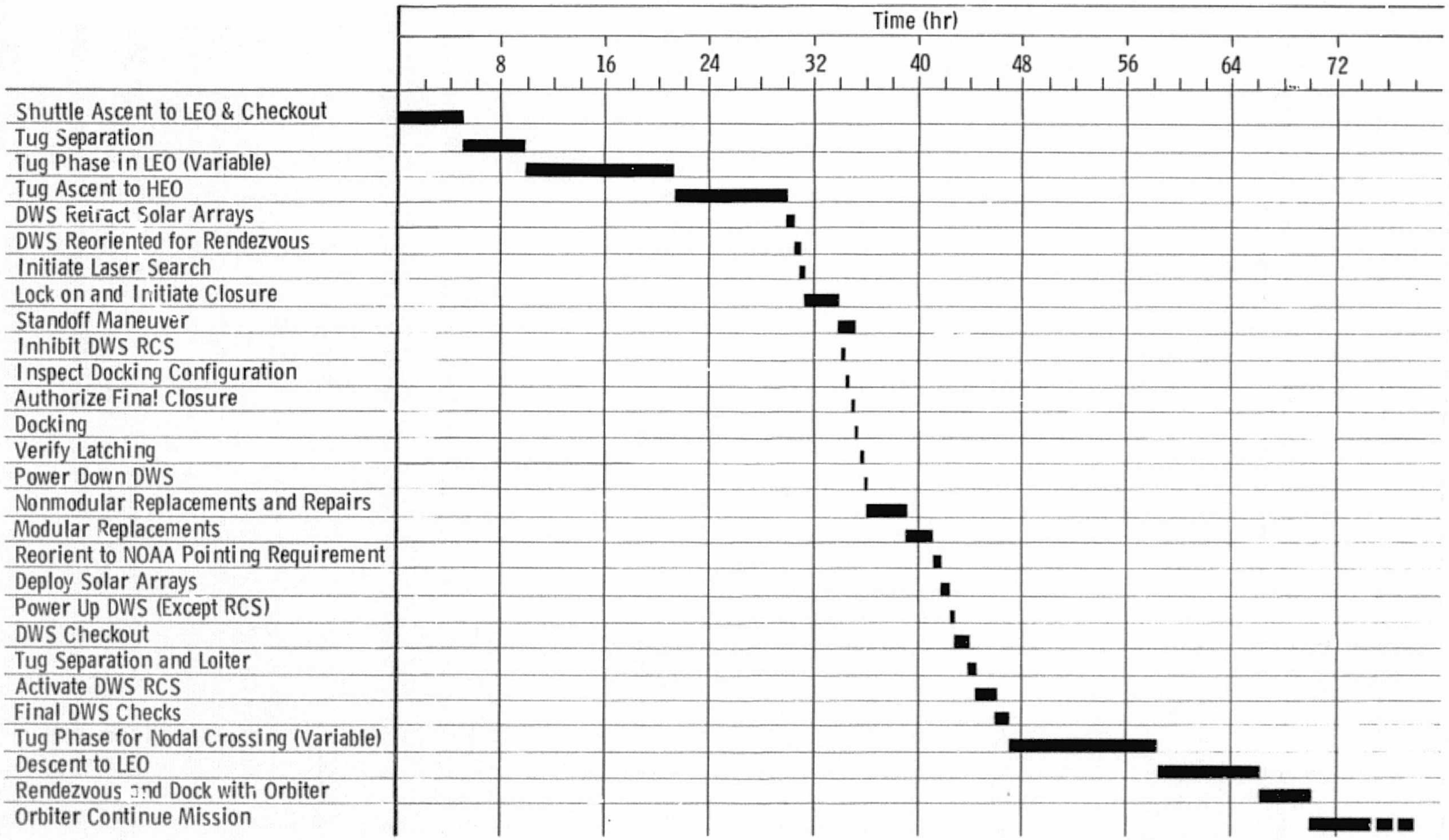


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Table IVD-4 Approach 1 - DWS Communications Paths

	Servicer/ Tug To DWS	Servicer/ Tug to STDN/MCC	Servicer/ Tug to TDRSS/MCC	MCC/STDN to Tug/ Servicer	MCC/TDRSS to Tug/ Servicer	MCC/STDN to Orbiter	MCC/TDRSS to Orbiter	NOAA to MCC to NOAA	MCC/STDN/ NOAA to DWS	DWS to NOAA
Shuttle Ascent to LEO & Checkout						X	X			
Tug Separation		X	X	X	X	X	X			
Tug Phase in LEO		X	X	X	X	X	X			
Tug Ascent to HEO		X	X							
DWS Retract Solar Arrays									X	
DWS Reoriented for Rendezvous									X	
Activate Laser Search	X-Radar	X								
Lock-on and Initiate Closure		X								
Standoff Maneuver		X								
Inhibit DWS RCS									X	
Inspect Docking Configuration	X - TV	X								
Authorize Final Closure				X						
Docking		X								
Verify Latching		X								
Power Down DWS	X									
Non-modular Replacements & Repairs		X								
Modular Replacements		X								
Reorient to NOAA Pointing Requirement		X		X				X		
Deploy Solar Arrays		X							X	
Power Up DWS (Except RCS)	X	X		X						
DWS Checkout		X						X	X	X
Tug Separation and Loiter		X		X						
Activate DWS RCS		X							X	
Final DWS Checks		X						X	X	X
Tug Phase for Nodal Crossing		X		X						
Descent to LEO		X	X	X	X					
Rendezvous and Dock with Orbiter			X		X		X			
Orbiter Continue Mission							X			

Table IVD-5 Approach 1 - DWS Power Sources

	ORBITER	TUG	DWS
Shuttle Ascent to LEO & Checkout	X		
Tug Separation		X	
Tug Phase in LEO		X	
Tug Ascent to HEO		X	
DWS Retract Solar Arrays			X
DWS Reoriented for Rendezvous			X
Initiate Laser Search		X	
Lock-on and Initiate Closure		X	
Standoff Maneuver		X	
Inhibit DWS RCS			X
Inspect Docking Configuration		X	
Authorize Final Closure			
Docking		X	
Verify Latching		X	
Power Down DWS		X	
Non-modular Replacements & Repairs		X	
Modular Replacements		X	
Reorient to NOAA Pointing Requirement		X	
Deploy Solar Arrays		X	
Power Up DWS (Except ACS)			X
DWS Checkout			X
Tug Separation and Loiter		X	
Activate DWS RCS			X
Final DWS Checks			X
Tug Phase for Nodal Crossing		X	
Descent to LEO		X	
Rendezvous and Dock with Orbiter	X	X	
Orbiter Continue Mission	X		

Table IVD-6 Approach 1 - DWS Control/Commands Sources

	MCC TO TUG/ SERVICER	MCC/NOAA TO DWS	TUG/SERVICER TO DWS
Shuttle Ascent to LEO & Checkout			
Tug Separation			
Tug Phase in LEO			
Tug Ascent to HEO	X		
DWS Retract Solar Arrays		X	
DWS Reoriented for Rendezvous		X	
Initiate Laser Search	X		
Lock-on and Initiate Closure			
Standoff Maneuver			
Inhibit DWS RCS		X	
Inspect Docking Configuration			
Authorize Final Closure	X		
Docking			
Verify Latching			
Power Down DWS			X
Non-modular Replacements & Repairs			
Modular Replacements			
Reorient to NOAA Pointing Requirement	X		
Deploy Solar Arrays			X
Power-up DWS (Except ACS)	X		
DWS Checkout		X	
Tug Separation and Loiter			
Activate DWW RCS		X	
Final DWS Checks		X	
Tug Phase for Nodal Crossing	X		
Descent to LEO	X		
Rendezvous and Dock with Orbiter			
Orbiter Continue Mission			

Tug

1) Provide computer preprogrammed control of the servicer mechanisms.
2) Relay of remote control commands to the servicer.
3) Relay of data from the DWS to the ground.
4) Relay of data from the servicer to the ground or to the orbiter during P/L bay checkouts.

b. Approach 2 - It is assumed the maintenance tasks to be performed are known. A manned servicing module capable of the maintenance tasks (modular changeout using EVA crewmen) is delivered to the DWS orbit by the use of two Tugs. The Tug/MSM assembly is docked to the DWS. The maintenance tasks are performed. The Tug/MSM is returned to the Shuttle Orbiter and subsequently to the ground.

The general mission sequence and primary operations would be:

1) MSM is transferred to the DWS orbit using two tandem Tugs.

2) Tug/MSM automatic rendezvous and docking.

3) Umbilical lockup with DWS (power, command, data, etc.).

4) Deactivate DWS systems.

5) Wait for removal of residual reaction wheel momentum (2 hours). Tug maintain attitude control.

6) Perform modular replacements or repairs by direct crew EVA.

7) Reorient assembly to functional attitude.

8) Deploy solar arrays. Power up DWS (except RCS).

9) Perform preliminary checkout.

10) When DWS reaction wheel speed is up, undock Tug/MSM and separate to safe distance and loiter.

11) Activate DWS RCS.

12) Ground verify proper DWS functional operation.

13) Tug/MSM return to Shuttle Orbiter.

Mission Details

Tug/MSM Transfer to DWS Orbit--Because of the weight of a manned module, a single Tug is not capable of delivering (and returning) the MSM to geosynchronous orbit. The weight breakdown for the MSM is presented in Table IVD-7.

Table IVD-7 MSM Weight Summary

Crew	400
ECLSS	500
Crew Systems	700
EPS	1,000
Expendables	700
Comm/data	150
Controls/displays	200
Structure	2,000
EVA Support Equipment	500
	6,150

Taken in part from SD73-SA-0036-4 and -5, Geosynchronous Platform Definition Study, Rockwell International, June 1973.

Tandem Tug delivery will be used. Two Shuttle flights will deliver the servicing assembly to Shuttle orbit. A Tug and Tug/Adapter will be delivered in the first Orbiter. A Tug and MSM will be delivered in the second Orbiter.

After preliminary checks using Shuttle systems, the two-man crew will enter the MSM. Means of entry need to be studied further. If an airlock transfer tunnel is used for shirtsleeve entry, some retraction mechanism would be needed. EVA entry might be more feasible. After entry, the MSM would be pressurized for shirtsleeve operation and final systems checks would be performed.

The Tug/MSM would then be assembled to the front of the second Tug in the adjacent Orbiter, using the RMS. The tandem assembly would then be deployed using the RMS. On separation to a safe distance, the Tug fuel cells will be started, all Tug systems activated, and functional readiness verified. At the proper place in the orbit, the first Tug Main Propulsion System (MPS) boost burn will be initiated to place the tandem assembly in a 160 x 7,000 n mi phasing orbit. During the first revolution, the two Tugs will separate. At about perigee, the first Tug will de-orbit to return to the Orbiter and the Tug/MSM MPS will

C-4

be fired to inject the Tug/MSM into a transfer orbit with the apogee at the geosynchronous altitude. At geosynchronous altitude the MPS is again burned to circularize the orbit of the Tug/MSM. Proper selection of the burn points in the phasing orbits will place the Tug/MSM near the DWS longitudinal location. The 28.5 deg plane change is accomplished by out-of-plane-vector firing, distributed at each burn. Most of the plane change is accomplished at the geosynchronous circularization burn.

Table IVD-8 presents the mission budgets. The ΔV budgets are not optimized for most efficient Tug utilization, but the propellant margin indicates the Tug boost capabilities are adequate. The maximum spares weight of 1,329 lbs was assumed (see approach 1).

Tug/MSM Rendezvous and Docking--The rendezvous and docking of the Tug/MSM with the DWS will be accomplished automatically for the most part. On acquisition of the DWS by laser radar, the Tug/MSM will initiate closure maneuvers. During the last phases of this period, the ground controllers will initiate commands to roll in the DWS solar arrays (see approach 1 for discussion concerning docking loads).

Prior to docking, the Tug/MSM will hold close to the DWS while the MSM crew will verify the predocking configuration. The DWS RCS will be inhibited by ground command to preclude any desaturation maneuvers at the time of docking. Docking will be completed automatically. However, the capability should exist for the MSM crewmen to assume control of the docking maneuvers. On docking, signals to the MSM will verify capture-latch closures.

Prepare DWS for Maintenance--On docking, the MSM crewmen will engage an umbilical connector and deactivate the DWS systems. The reaction wheel momentum will be removed in about two hours by use of motor braking.

During this and subsequent periods, the Tug will maintain attitude control of the Tug/MSM/DWS assembly. The attitude will be such that adequate Tug communications pointing is maintained as well as DWS orientation for thermal control to prevent excessively low temperatures.

Perform Maintenance--The two MSM crewmen would now proceed to the maintenance tasks by extravehicular activity (EVA). Because of the limitations on work time from the crewmen, the time required for EVA preps, and the time for post-EVA securing of equipment, two EVAs were assumed for the maintenance tasks and a total time of 48 hours (including sleep periods) was assumed.

Both crewmen would don suits. The MSM would be depressurized. One crewman would perform the actual maintenance tasks. The second crewman would transfer spares back and forth from a station near the EVA hatch, using Skylab-type transfer booms.

Table IVD-8 Approach 2 - DWS Geosynchronous Mission Budgets

Event	Duration (hours)	Inerts/* Losses (lbs) A	Inerts/* Losses (lbs) B	APS (lbs) A	APS (lbs) B	MPS ΔV (ft/sec)	Initial Vehicle Weight (lbs)
First Shuttle/Tug Launch and Wait	48.0	144.0					120,574
Second Shuttle/Tug/MSM Launch	2.0	6.0					120,430
MSM Checkout	1.0	3.0	3.0				120,418
Tug Mating	1.0	3.0	3.0				120,412
Separation and Checkout	3.0	10.0	10.0	10.0			120,406
Phase in Shuttle Orbit (160 n mi)	11.0	46.0	46.0	30.0			120,376
Phasing/Plane Change (2°) Burn	0.35					5950	120,254
Coast 1 Rev. in Phasing Orbit	5.0	20.0	20.0	10.0	10.0		79,875
Tugs Separate							79,815
Tug A Deboost for Return	0.08					5950	15,797
Rendezvous and Docking	4.0			32.0			10,493
Tug A: Propellant Consumed, 45,683 lbs; Propellant reserve 4,206; APS used 82; APS reserve 29; Tug adapter 200							10,461
Tug B Inject into Geosynchronous Transfer (perigee burn and 2-1/2° plane change	0.10					2133	64,018
Coast to Midcourse Correction	1.5		6.0		14.0		55,284
Midcourse Correction	0.03					50	55,264
Coast to 19,300 n mi Apogee	3.96		16.0		14.0		55,038
Circularize and Plane Change (24°) at Geosynchronous Altitude	0.12					5895	55,008
Rendezvous and Docking	6.0		24.0		97.0		36,676
On-Orbit Maintenance	56.0		224.0		40.0		36,555
Phase at Geosynch for Nodal Crossing	11.4		45.0		11.0		36,291
Deboost Burn	0.08					5840	36,235
Coast to Midcourse Correction	1.0		6.0		8.0		24,251
Midcourse Correction	0.01					13	24,237
Coast to 170 n mi Perigee	4.2		24.0		8.0		24,211
Inject into Return Phasing	0.05					3791	24,179
Coast 1 Rev. in Phasing Orbit	3.0		18.0		8.0		18,631
Circularize at 170 x 170 n mi orbit	0.05					4243	18,605
Rendezvous and Docking	4.0				32.0		13,897
TOTALS		232.0	451.0	82.0	242.0		

Tug B:
- Propellant Consumed, 49,558 lbs
- Propellant Unused Capacity 631
- APS used 242
- APS unused capacity 75

Returned Weight		13,865
Tug dry weight	5150	
Unusable Residuals	576	
APS reserve	29	
Propellant reserve	631	
Spares	1329	
MSM**	6150	
	13,865	

* Boil-off, fuel cell consumables.

** No life support system depletion - worst return case

EVA offers the greatest versatility for performing maintenance tasks. DWS equipment should still be grouped and installed in replaceable modular units as much as possible. This simplifies the tasks to be performed. In addition to modular replacements, the EVA crewman could perform repair-type tasks, such as mechanical splicing and taping of wires, taping of holes or rips in the antenna, and straightening of bends in structures.

Contingencies--Communications will be maintained between the crewmen and the ground at all times. In the event problems are encountered in successfully accomplishing the maintenance tasks, malfunctions occur that jeopardize safety of equipment, or normal undocking is not possible, the ground controllers will make real-time decisions on corrective action. For example, if some maintenance is required which the on-orbit MSM might not be capable of, but a subsequent one could perform depending on Shuttle/Tug availabilities, the decision might be made to continue the planned tasks. The remaining maintenance tasks would be accomplished with a subsequent mission. The subsequent Tug/MSM would need less spares for the DWS but the mission could be combined with maintenance, delivery, or retrieval of some other satellite. This would eliminate an additional mission totally dedicated to the one DWS.

The single point failure of a capture-latch failing to release, or a similar failure, could prevent recovery of the Tug. Corrective action could probably be achieved by EVA; however, means should be provided to mechanically (and/or by pyrotechnics) separate the Tug/MSM at the DWS interface.

Reactivate DWS Systems--On completion of all maintenance tasks, the solar arrays would be deployed and all DWS systems except the RCS would be reactivated. The assembly would be reoriented to the normal DWS pointing attitude. Preliminary checkouts would be performed to verify DWS mission performance. Since the disaster warning network is separate from the NASA network, the NOAA personnel would perform the operational checks in conference with the NASA Tug/SC controllers.

Separate and Perform Final Checks--When the reaction wheel speed is up, the Tug/MSM will undock and separate to a safe distance, and the DWS RCS will be activated. The Tug will loiter in close proximity until all system checks are verified by NOAA and NASA controllers.

Tug/MSM Return to Orbiter--The deboost burn and return to the orbiter will be accomplished per the budget and schedule

presented in Table IVD-8. Final rendezvous and docking using the RMS will be accomplished by the orbiter crew. The MSM crew will reenter the Orbiter cabin before Shuttle return to ground.

Discussion

As seen in Table IVD-8, the DWS maintenance mission, using a 6,150 lb manned module and carrying 1,329 lbs of spares, requires two tandem tugs. Considerable propellant reserve capability exists, however, permitting greater mission times and/or Tug payload weight. Greater manned module life support consumables could be accommodated to permit longer in-orbit time.

The mission timeline is presented in Table IVD-9. A period of 48 hours is assumed for the time to launch a second Shuttle after the first launch. It is noted that the Shuttle orbiters will be free to perform other mission operations except for the times of Tug deployments and recaptures.

The most serious safety aspect of this approach is the possibility of Tug propulsion failure and the resultant loss of capability to return the MSM to the orbiter. A rescue mission would be required. This would require another Shuttle/Tug to be prepared for launch. Assuming preparations for this third launch are started after launch of the second Shuttle, the third vehicle would be ready for launch before the geosynchronous maintenance operations are complete. Assuming the propulsion failure to be found when ready to deboost the Tug/MSM, the rescue mission could be effected in a time approximately 48 hours past the time the MSM would normally be returned to the orbiter. This would not require an excessive amount of additional life support consumables and fuel cell reactants. Provisions would be needed to permit undocking the MSM from the disabled Tug and redocking to the rescue Tug. Some attitude control capability would be needed to stabilize the MSM from any undocking perturbations.

This approach would require man-rated Tugs and DWS. To be compatible with EVA, the DWS would need tether attachment points, foot restraints, hand-holds, and translation rails. Additional DWS fabrication precautions would be needed to assure no sharp edges and protrusions in areas of possible EVA.

Table IVD-10 details the communications paths during the mission. During LEO operations, the TDRSS will be used. The STDN will be used during geosynchronous orbit operations. No communications incompatibilities are noted.

Table IVD-9 Approach 2 - DWS Mission Timelines

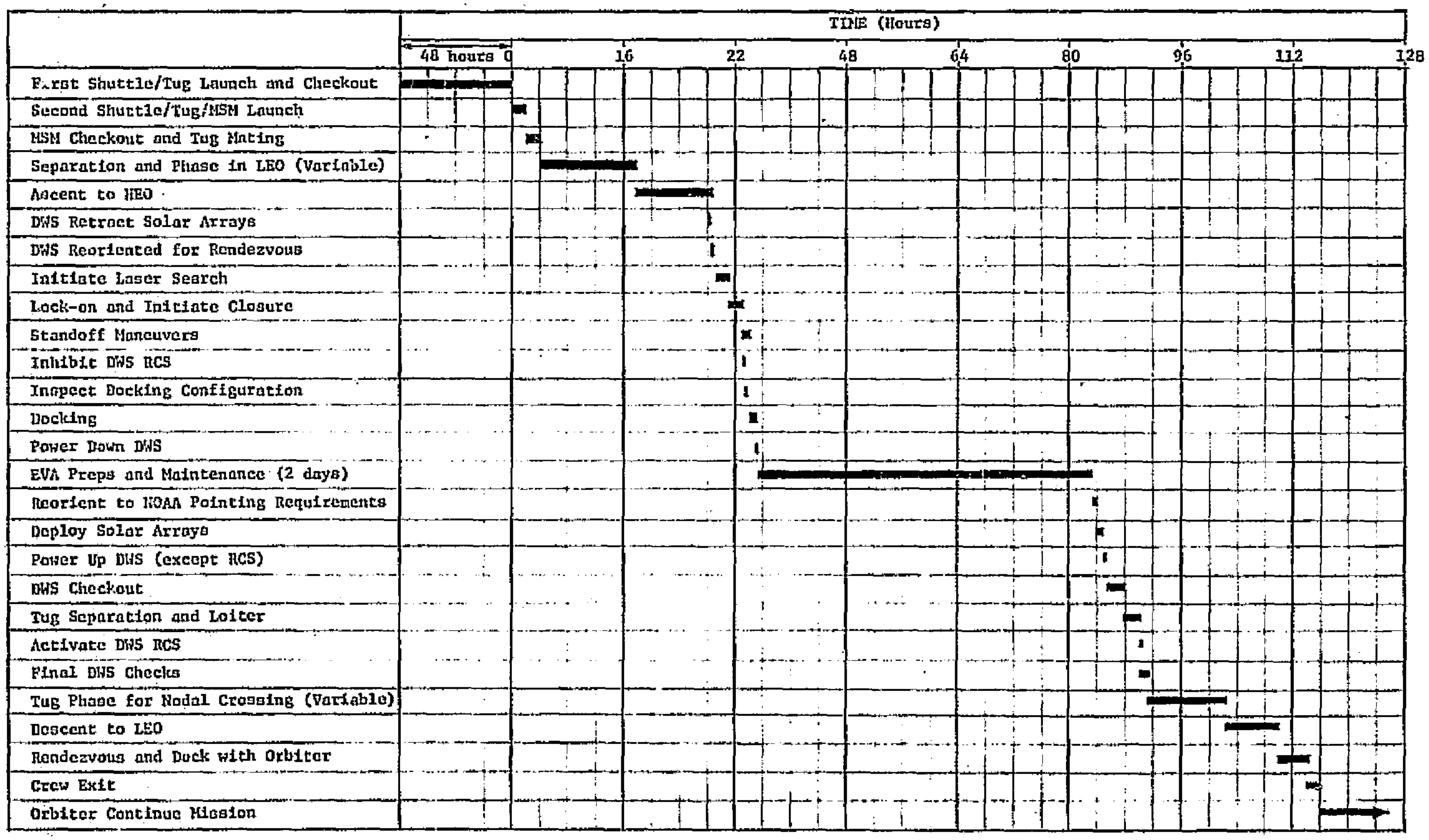

Table IVD-10 Approach 2 - DWS Communications Paths

	MSM/Tug to DWS	MSM/Tug to STDN/MCC	MSM/Tug to TDRSS/MCC	MCC/STDN to Tug/MSM	MCC/TDRSS to Tug/MSM	MCC/STDN to Orbiter	MCC/TDRSS to Orbiter	NOAA to MCC to NOAA	MCC/STDN/NOAA to DWS	DWS to NOAA
First Shuttle/Tug Launch and Checkout						X	X			
Second Shuttle/Tug/MSM Launch						X	X			
MSM Checkout and Tug Mating		X	X	X	X	X	X			
Separation and Phase in LEO		X	X	X	X	X	X			
Ascent to HEO		X	X	X	X					
DWS Retract Solar Arrays									X	
DWS Reoriented for Rendezvous									X	
Initiate Laser Search	X-Radar	X								
Lock-on and Initiate Closure		X								
Standoff Maneuvers		X								
Inhibit DWS RCS									X	
Inspect Docking Configuration	X	X								
Docking		X								
Power Down DWS	X									
EVA Preps and Maintenance (2 days)		X		X						
Reorient to NOAA Pointing Requirements		X		X						
Deploy Solar Arrays	X									
Power Up DWS (except RCS)	X									
DWS Checkout		X		X						X
Tug Separation and Loiter		X		X						
Activate DWS RCS		X							X	
Final DWS Checks		X		X						X
Tug Phase for Nodal Crossing		X		X						
Descent to LEO		X	X	X	X					
Rendezvous and Dock with Orbiter			X		X		X			
Crew Exit							X			
Orbiter Continue Mission							X			

Table IVD-11 presents the electrical power sources. The baseline Tug has the capability to supply 600 watts to a payload. The power required by the MSM during the maintenance operations shall be limited to 600 watts. In the event of Tug electrical power failure, the MSM must be capable of providing battery power until the return to the orbiter or until rescue is effected.

Table IVD-12 summarizes the various commands during the mission. The capability is needed to control DWS subsystem functions from the MSM via the umbilical connector and from the ground via telemetry. The capability to maneuver the Tug from the MSM is also needed.

The following capabilities are required because of this approach to maintain the DWS:

<u>DWS</u>

1) Capability to roll in and deploy the solar arrays.
2) Receptacle for umbilical attachment from the MSM.
3) Laser radar reflectors (corner cubes).
4) Replaceable subsystem modules, including solar arrays.
5) Docking frame, probe, and latches compatible with the MSM.
6) Solar array booms capable of withstanding docking loads.
7) Redundant shutoffs, structural safety factors, no sharp edges or protrusions, and similar aspects to minimize hazards to EVA crewmen.

<u>MSM</u>

1) Docking provisions compatible with DWS.
2) Life support systems for IVA and EVA.
3) Umbilical system capable of conveying control commands and electrical power to the DWS.
4) Backup means of separation from the DWS in the event of failure of capture-latches to open.
5) Capability to control Tug APS maneuvers.
6) Contingency life support and electrical power reserves.
7) Transmission of life support system status to ground through the Tug.
8) Attitude control capability to stabilize MSM for docking with rescue Tug.
9) Provisions for undocking/docking with the Tug in orbit. Includes laser reflectors.
10) Provisions for general flood lighting in the DWS subsystem modules area and for portable lighting in other maintenance areas.

Table IVD-11 Approach 2 - DWS Power Sources

	Orbiter	Tug	DWS
First Shuttle/Tug Launch and Checkout	X		
Second Shuttle/Tug/MSM Launch	X		
MSM Checkout and Tug Mating	X		
Separation and Phase in LEO		X	
Ascent to HEO		X	
DWS Retract Solar Arrays			X
DWS Reoriented for Rendezvous			X
Initiate Laser Search		X	
Lock-on and Initiate Closure		X	
Standoff Maneuvers		X	
Inhibit DWS RCS			X
Inspect Docking Configuration		X	
Docking		X	
Power Down DWS		X	
EVA Preps and Maintenance (2 days)		X	
Reorient to NOAA Pointing Requirements		X	
Deploy Solar Arrays		X	
Power Up DWS (except RCS)			X
DWS Checkout			X
Tug Separation and Loiter		X	
Activate DWS RCS			X
Final DWS Checks			X
Tug Phase for Nodal Crossing		X	
Descent to LEO		X	
Rendezvous and Dock with Orbiter	X	X	
Crew Exit	X		
Orbiter Continue Mission	X		

Table IVD-12 Approach 2 - DWS Commands

	MCC to Tug/MSM	MCC/NOAA to DWS	Tug/MSM to DWS	MSM/Tug
First Shuttle/Tug Launch and Checkout				
Second Shuttle/Tug/MSM Launch				
MSM Checkout and Tug Mating				
Separation and Phase in LEO				
Ascent to HEO	X			
DWS Retract Solar Arrays		X		
DWS Reoriented for Rendezvous		X		
Initiate Laser Search				X
Lock-on and Initiate Closure				X
Standoff Maneuvers				X
Inhibit DWS RCS		X		X
Inspect Docking Configuration				X
Docking				X
Power Down DWS			X	
EVA Preps and Maintenance (2 days)				X
Reorient to NOAA Pointing Requirements	X			
Deploy Solar Arrays			X	
Power Up DWS (except RCS)			X	
DWS Checkout		X		
Tug Separation and Loiter				X
Activate DWS RCS		X		
Final DWS Checks		X		
Tug Phase for Nodal Crossing	X			
Descent to LEO	X			
Rendezvous and Dock with Orbiter				
Crew Exit				
Orbiter Continue Mission				

11) Provisions for transferring spares to and from maintenance areas from MSM hatch.

Tug

1) Compatibility with manned mission (man-rated).
2) Relay of data from the DWS/MSM to the ground.
3) Relay of communications between crewmen and ground.
4) Relay of data and communications between the MSM/Tug and the orbiter during P/L bay checkouts.
5) Provisions for docking with MSM in orbit. Laser radar capability with or without MSM attached.

Orbiter

1) Provisions for attaching airlock transfer tunnel to MSM and retracting tunnel.

c. Approach 3 - In this approach, the DWS is retrieved from geosynchronous orbit by a Tug and returned to the Shuttle Orbiter. Maintenance is performed at the Orbiter using the RMS and EVA crewmen. The DWS is then replaced in geosynchronous orbit by another Tug launched by another Shuttle.

The general mission sequence and primary operations would be:

1) Tug transfer to DWS orbit.

2) Inhibit DWS RCS and roll in solar arrays.

3) Tug automatic rendezvous and docking with the DWS.

4) Engage umbilical connector.

5) Deactivate DWS systems and fold antenna.

6) Tug/DWS return to Shuttle Orbiter.

7) Maintenance performed by RMS and EVA crewmen.

8) Functional checkout of DWS.

9) Attach DWS to loaded Tug launched in second Shuttle.

10) Tug transfer DWS to geosynchronous orbit.

11) Ground verify proper DWS function.

12) Tug return to Shuttle Orbiter.

Mission Details

Retrieval of DWS--The retrieving Tug will be deployed from the Shuttle P/L bay, activated, checked out, and launched to the DWS orbit. Table IVD-13 presents the mission and ΔV budgets for the retrieval mission.

The rendezvous and docking of the Tug to the DWS will be accomplished automatically. During the last phases of this period, the ground will command the solar arrays to be rolled in to minimize bending loads on the array booms. On docking, signals will verify capture-latch closures. An umbilical connector will engage the DWS and deactivate all DWS systems. The Tug will maintain attitude control for thermal control of the DWS and pointing for Tug communications with the ground.

Deceleration will approach 2 g's during the return to the Shuttle. The SSPD calls for a maximum acceleration of 0.1 g when appendages are deployed. However, the SSPD also indicates the DWS would be transferred from LEO to HEO with appendages deployed. In this case, acceleration would be over 0.5 g. It is doubtful that the SSPD is accurate in these concerns. Regardless, the DWS must be designed either to provide means for folding the appendages (solar arrays and 19-foot antenna) prior to transfer, or be structurally capable to accommodate acceleration of 2 g's. It is assumed herein that the antenna is folded prior to transfer.

On rendezvous with the Shuttle Orbiter, the RMS would dock the Tug and attach it at the P/L bay aft tilt table.

Perform Maintenance--Module replacements and repair tasks would be accomplished by EVA crewmen. The EVA crewman would be transferred to the DWS by the RMS. The EVA crewman would position himself in foot restraints at the DWS work-site. Use of portable foot restraints would alleviate the need for permanent restraints on the DWS and save on satellite weight. Spare modules would be transferred by the RMS from storage in the

Table IVD-13 Approach 3 - DWS Geosynchronous Retrieval Mission Budgets

Event	Duration Hours	Inerts/* Losses (Lbs)	APS (Lbs)	MPS ΔV (Ft/Sec)	Initial Vehicle Weight (Lbs)
Tug Separation from Orbiter	2.0	10.0	8.6		56625
Phase in Shuttle Orbit (160 n.mi.)	11.0	46.0	21.4		56608
Phasing/Plane Change Burn	0.13			4494	56541
Coast 1 Rev. in Phasing Orbit	3.0	5.0	17.5		41510
Inject into Geosynchronous Transfer (Perigee Burn)	0.11			3672	41487
Coast to Midcourse Correction	1.5	6.0	13.8		32229
Midcourse Correction	0.03			50	32209
Coast to 19,300 n.mi. Apogee	3.46	16.0	14.0		32078
Circularize at Geosynchronous Altitude (Apogee Burn)	0.12			5828	32048
Rendezvous and Docking	4.0	16.0	96.5		21466
Phase in Orbit for Nodal Crossing	11.4	45.0	11.2		23257
Deboost Burn	0.08			5840	23201
Coast to Midcourse Correction	1.0	6.0	7.5		15528
Midcourse Correction	0.02			35	15514
Coast to 170 n.mi. Perigee	4.2	24.0	8.1		15470
Inject into Return Phasing Orbit	0.05			3791	15438
Coast 1 Rev. in Phasing Orbit	3.0	18.0	7.8		11895
Circularize at 170 x 170 n.mi. Orbit	0.05			4243	11869
Rendezvous and Dock with Orbiter	4.0		32.4		8866
TOTALS	49.2	192.0	238.8		

Propellant Consumed	49265
Unused Propellant Capacity	924
APS Consumed	239
Unused APS Capacity	78

<u>Returned Weight</u>		<u>8833</u>
Tug Dry Weight	5150	
Unusable Residuals	576	
APS Reserve	29	
Propellant Reserve	924	
DWS Docking Frame	250	
DWS	<u>1904</u>	
	8833	

*Boiloff, fuel cell consumables

P/L bay. The second EVA crewman would assist operations from a position in the P/L bay. During the maintenance phase, communications will be maintained between the EVA crewmen and other crewmen in the Orbiter cabin via RF. Communications with ground controllers will be maintained through the Orbiter systems and the TDRSS.

Functional Checkout--After the maintenance tasks are completed, the DWS will be pointed at the selected spot on earth by Shuttle maneuvers. The DWS antenna and solar arrays will be deployed and the DWS systems (except ACS) will be powered up. (See Figure IVD-7). Functional checks would then be performed by NOAA personnel, in conference with NASA ground controllers. Because of the low orbit and the 28.5 deg inclination of the Orbiter, the functional checks will be performed on long passes over the U. S. Some Shuttle maneuvering will be required to maintain the proper DWS antenna pointing during the passes.

DWS Transfer to Second Tug--After all functional verification checks are completed, the DWS would be powered down and the appendages would again be rolled in. When the second Shuttle/Tug is launched and in the appropriate stationkeeping position, the DWS will be transferred to the loaded Tug, using the RMS from both orbiters.

Transfer of DWS to Geosynchronous Orbit--The DWS would then be tranferred to geosynchronous orbit and placed in the proper longitudinal position. See Table IVD-14 for the transfer budgets and schedule. The Tug would deploy the DWS and loiter in the vicinity until final verification from NOAA that all systems are functioning. The Tug would then return to the Shuttle Orbiter.

Discussion

As seen in the Tug budgets and the mission timelines (Table IVD-15) no constraints on time or boost capabilities are foreseen. The Shuttle at which DWS maintenance is performed is free for other mission operations about 4-1/2 days out of the nominal 7-day mission time. The second Shuttle is free for other operations all of the time except for about 10 hours.

Table IVD-16 details the communications paths during the mission. No communications incompatibilities are foreseen.

Table IVD-17 presents the electrical power sources. DWS power will be required from the Tug and Shuttle for the deployment and folding of the appendages.

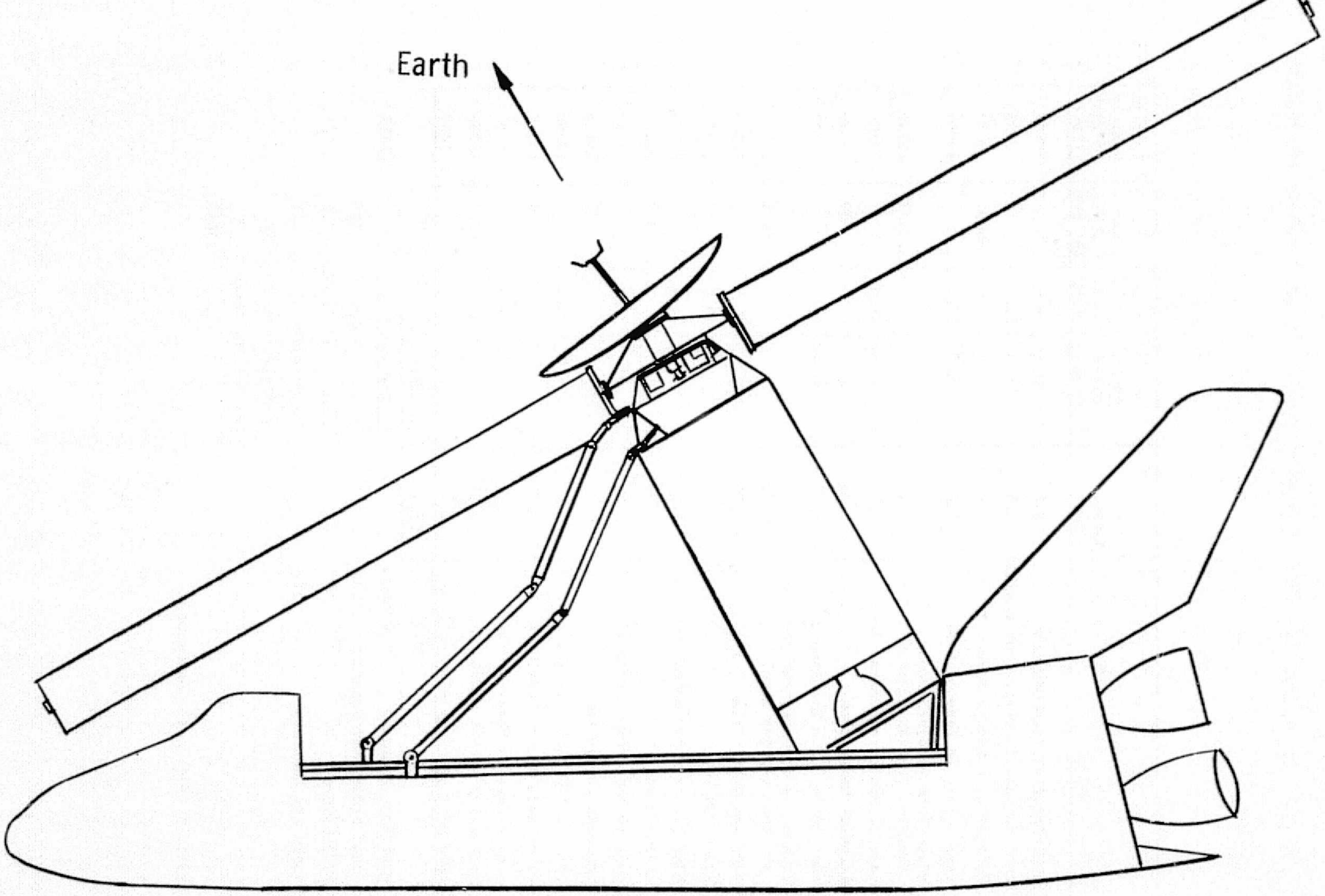

Figure IVD-7 DWS Functional Checkout Attitude

Table IVD-14 Approach 3 - DWS Geosynchronous Delivery Mission Budgets

Event	Duration Hours	Inerts/* Losses (Lbs)	APS (Lbs)	MPS ΔV (Ft/Sec)	Initial Vehicle Weight (Lbs)
Tug Separation from Orbiter	2.0	10.0	8.6		58605
Phase in Shuttle Orbit (160 n.mi.)	11.0	46.0	21.4		58586
Phasing/Plane Change Burn	0.13			4494	58519
Coast 1 Rev. in Phasing Orbit	3.0	5.0	17.5		42963
Inject into Geosynchronous Transfer (Perigee Burn)	0.11			3672	42940
Coast to Midcourse Correction	1.5	6.0	13.8		33358
Midcourse Correction	0.03			50	33338
Coast to 19,300 n.mi. Apogee	3.46	16.0	14.0		33202
Circularize at Geosynchronous Altitude (Apogee Burn)	0.12			5828	33172
Coast and Orbit Trim	12.0	48.0	96.5		22219
Deploy DWS	1.0	4.0	40.0		22074
Phase in Orbit for Nodal Crossing	11.4	45.0	11.2		20126
Deboost Burn	0.08			5840	20070
Coast to Midcourse Correction	1.0	6.0	7.5		13432
Midcourse Correction	0.02			35	13418
Coast to 170 n.mi. Perigee	4.2	24.0	8.1		13380
Inject into Return Phasing Orbit	0.05			3791	13348
Coast 1 Rev. in Phasing Orbit	3.0	18.0	7.8		10285
Circularize at 170 x 170 n.mi.	0.05			4243	10259
Rendezvous and Dock with Orbiter	4.0		32.4		7663
TOTALS	58.3	228.0	278.8		

Propellant Consumed	48562.	Returned Weight	7632
Unused Propellant Capacity	1627.	Tug dry weight	5150
APS Consumed	279.	Unusable residuals	576
Unused APS Capacity	38.	APS reserve	29
		Propellant reserve	1627
		DWS Docking Frame	250
			7632

*Boiloff, fu[illegible] [illegible] consumables

Table IVD-15 Approach 3 - DWS Mission Timeline

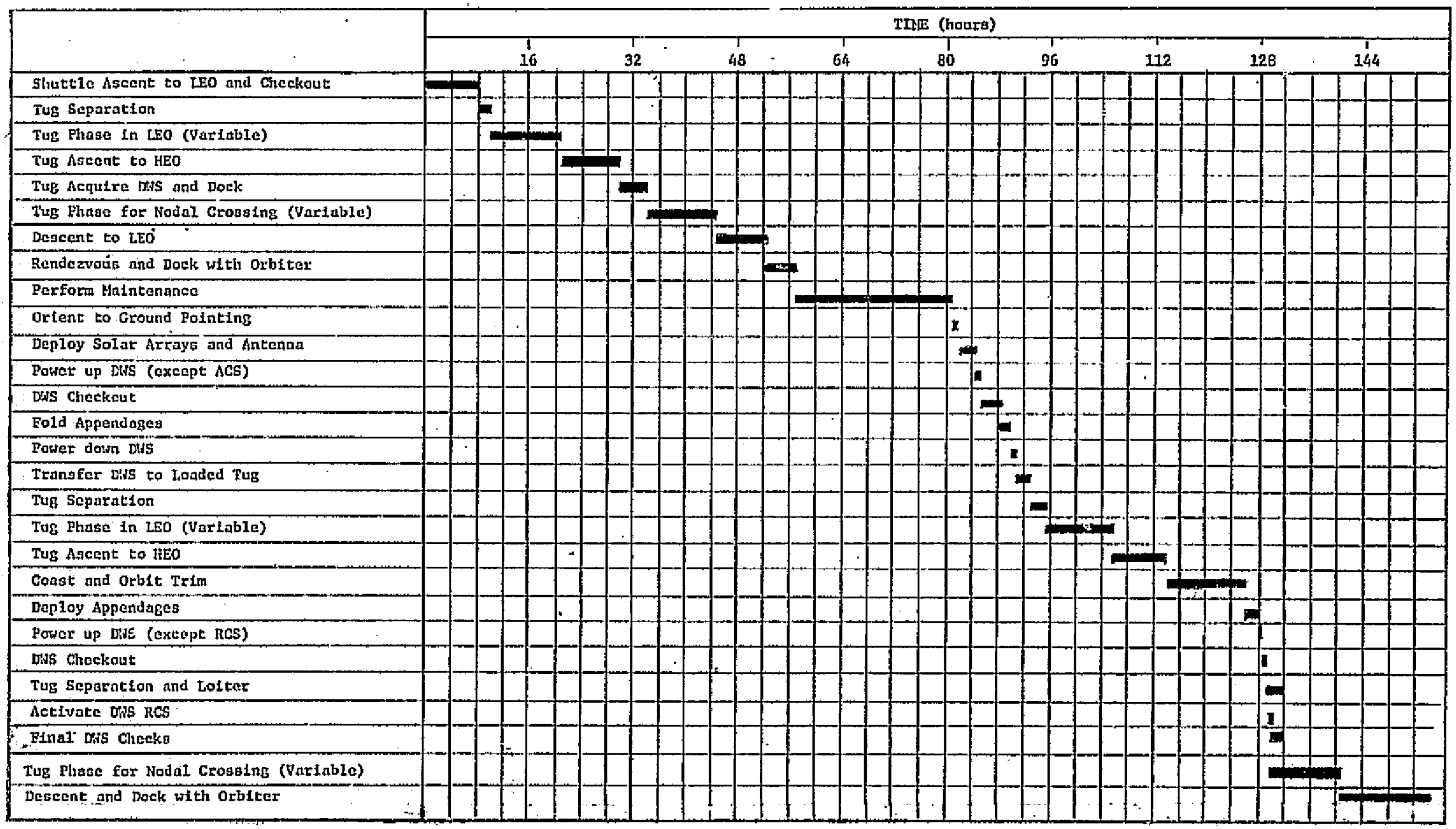

Table IVD-16 Approach 3 - DWS Communications Paths

	Tug to DWS	Tug to STDN/MCC	Tug to TDRSS/MCC	MCC/STDN to Tug	MCC/TDRSS to Tug	MCC/STDN to Orbiter	MCC/TDRSS to Orbiter	NOAA to MCC to NOAA	MCC/STDN/ NOAA to DWS	DWS to NOAA
Shuttle Ascent to LEO and Checkout						X	X			
Tug Separation			X		X	X	X			
Tug Phase in LEO (Variable)			X		X					
Tug Ascent to HEO		X	X	X	X					
Tug Acquire DWS and Dock	X	X		X						
Tug Phase for Nodal Crossing		X		X						
Descent to LEO		X	X	X	X					
Rendezvous and Dock with Orbiter			X		X		X			
Perform Maintenance							X			
Orient to Ground Pointing							X	X		
Deploy Solar Arrays and Antenna							X	X		
Power up DWS (except ACS)							X	X		
DWS Checkout							X	X		X
Fold Appendages							X	X		
Power down DWS							X	X		
Transfer DWS to Loaded Tug			X		X		X			
Tug Separation			X		X					
Tug Phase in LEO (Variable)			X		X					
Tug Ascent to HEO		X	X	X	X					
Coast and Orbit Trim		X		X						
Deploy Appendages								X	X	X
Power up DWS (except RCS)	X							X	X	X
DWS Checkout								X	X	X
Tug Separation and Loiter		X		X						
Activate DWS RCS									X	
Final DWS Checks								X	X	X
Tug Phase for Nodal Crossing		X		X						
Descent and Dock with Orbiter		X	X	X	X					

Table IVD-17 Approach 3 - DWS Power Sources

	Orbiter	Tug	DWS
Shuttle Ascent to LEO and Checkout	X		
Tug Separation		X	
Tug Phase in LEO (Variable)		X	
Tug Ascent to HEO		X	
Tug Acquire DWS and Dock		X	
Tug Phase for Nodal Crossing (Variable)		X	
Descent to LEO		X	
Rendezvous and Dock with Orbiter	X	X	
Perform Maintenance	X		
Orient to Ground Pointing	X		
Deploy Solar Arrays and Antenna	X		
Power up DWS (except ACS)			X
DWS Checkout			X
Fold Appendages			X
Power down DWS	X		
Transfer DWS to Loaded Tug	X		
Tug Separation		X	
Tug Phase in LEO (Variable)		X	
Tug Ascent to HEO		X	
Coast and Orbit Trim		X	
Deploy Appendages		X	
Power up DWS (except RCS)			X
DWS Checkout			X
Tug Separation and Loiter		X	
Activate DWS RCS			X
Final DWS Checks			X
Tug Phase for Nodal Crossing (Variable)		X	
Descent and Dock with Orbiter		X	

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Table IVD-18 summarizes the various commands during the mission. The capability is needed to control DWS subsystem functions from the ground via telemetry and from the Tug and Shuttle systems via the umbilical connection.

The following additional capabilities are required because of this approach to maintenance of the DWS:

DWS

1) Capability to roll in and deploy the solar arrays.
2) Receptacle for umbilical attachment from the Tug.
3) Laser radar reflectors (corner cubes).
4) Replaceable subsystem modules, including solar arrays.
5) Docking frame probe and latches compatible with the Tug.
6) Solar array booms capable of withstanding docking loads.
7) Capability for multiple folding and deployment of the 19-foot antenna.
8) Redundant shutoffs, structural safety factors, no sharp edges or protrusions, and provisions for mounting portable foot restraints.

Tug

1) Relay of data between the DWS and the ground.
2) Relay of commands and power from the Shuttle to the DWS.
3) Provisions for docking with the DWS.

Shuttle Orbiter

1) Provide portable foot restraints.
2) Provide portable lights.

Applications of the three maintenance approaches to the other subject satellites were analyzed. Because of similarities to the DWS maintenance analysis, only differences in the maintenance requirements for the following satellites will be discussed.

Table IVD-18 Approach 3 - DWS Control/Commands Sources

	MCC to Tug	MCC/NOAA to DWS	Tug to DWS	Orbiter Crew
Shuttle Ascent to LEO and Checkout				X
Tug Separation	X			
Tug Phase in LEO (Variable)	X			
Tug Ascent to HEO	X			
Tug Acquire DWS and Dock	X	X	X	
Tug Phase for Nodal Crossing (Variable)	X			
Descent to LEO	X			
Rendezvous and Dock with Orbiter	X			X
Perform Maintenance				X
Orient to Ground Pointing				X
Deploy Solar Arrays and Antenna				X
Power up DWS (except ACS)				X
DWS Checkout		X		
Fold Appendages				X
Power down DWS				X
Transfer DWS to Loaded Tug				X
Tug Separation	X			X
Tug Phase in LEO (Variable)	X			
Tug Ascent to HEO	X			
Coast and Orbit Trim	X			
Deploy Appendages		X		
Power up DWS (except RCS)			X	
DWS Checkout		X		
Tug Separation and Loiter	X			
Activate DWS RCS		X		
Final DWS Checks		X		
Tug Phase for Nodal Crossing (Variable)	X			
Descent and Dock with Orbiter	X			X

3. Intelsat Maintenance

a. Approach 1 - The mission budgets for Approach 1 maintenance of the Intelsat are presented in Table IVD-19. Based on a servicer weight of 1,150 lbs, the Tug capability results in an allowable servicing spares weight of 1,200 lbs, or about 60% of the total replaceable modules weight.

Table IVD-19 Approach 1 - Intelsat Geosynchronous Mission Budgets

Event	Duration (hrs)	Inerts/* Losses (lbs)	APS (lbs)	MPS ΔV (ft/sec)	Initial Vehicle Weight (lbs)
Shuttle Ascent and Tug Preseparation	3.0	24.0			58,786
Tug Separation from Orbiter	2.0	10.0	8.6		58,762
Phase in Shuttle Orbit (160 n mi)	11.0	46.0	21.4		58,743
Phasing/Plane Change Burn	0.13			4494	58,676
Coast 1 Rev. in Phasing Orbit	3.0	5.0	17.4		43,076
Inject into Geosynchronous Transfer (Perigee Burn)	0.11			3672	43,054
Coast to Midcourse Correction	1.5	6.0	13.8		33,446
Midcourse Correction	0.03			50	33,426
Coast to 19,300 n mi Apogee	3.96	16.0	14.0		33,289
Circularize at Geosynchronous Altitude (Apogee Burn)	0.12			5826	33,259
Rendezvous and Docking	4.0	15.0	96.5		22,280
On-orbit Maintenance and Checkout	13.5	54.0	8.0		22,168
Phase at Geosynch for Nodal Crossing	11.4	45.0	11.2		22,106
Deboost Burn	0.08			5840	22,050
Coast to Midcourse Correction	1.0	6.0	7.5		14,757
Midcourse Correction	0.01			13	14,743
Coast to 170 n mi Perigee	4.2	24.0	8.1		14,727
Inject into Return Phasing	0.05			3791	14,695
Coast 1 Rev. in Phasing Orbit	3.0	18.0	7.8		11,322
Circularize at 170 x 170 n mi Orbit	0.05			4243	11,297
Rendezvous and Dock with Orbiter	4.0		32.4		8,438
TOTALS	63.1	245.0	246.8		

Propellant Consumed	49,865
Unused Propellant Capacity	324
APS Consumed	247
Unused APS Capacity	70

Returned Weight		8,405
Tug dry weight	5,150	
Unusable residuals	576	
APS reserve	29	
Propellant reserve	300	
Servicer	1,150	
Used spares	1,200	
	8,405	

*Boil off, fuel cell consumables.

Timelines, communications, power sources, and commands required for maintenance of the Intelsat would be very similar to those for the DWS. No incompatibilities are foreseen. The following capabilities are required because of this approach to maintenance of the Intelsat:

Intelsat

1) Capability to retract or deploy the solar arrays;
2) Receptacle for umbilical attachment from the servicer;
3) Laser radar reflectors (corner cubes);
4) Replaceable subsystem modules, including solar arrays;
5) Docking frame and latches compatible with the servicer;
6) Solar array booms capable of withstanding docking loads.

Servicer

1) Docking frame compatible with Intelsat;
2) Servicing system controlled by commands from Tug;
3) Servicing system capable of preprogrammed changeout of Intelsat modules and remote-control changeout of replaceable units. The latter will incorporate the use of TV;
4) Umbilical system capable of being connected to the Intelsat to convey control commands and electrical power;
5) Backup means of separation in the event capture latches fail to open.

Tug

1) Provide computer preprogrammed control of the servicer mechanisms;
2) Relay of remote control commands to the servicer;
3) Relay of data from the Intelsat to the ground;
4) Relay of data from the servicer to the ground or to the orbiter during P/L bay checkouts.

b. Approach 2 - Table IVD-20 presents the mission budgets for this approach. The analysis is based on the first Tug boosting the assembly into a 160 x 11,000 n mi phasing orbit. The approach requires two Tugs but presents more-than-adequate capability to carry a full complement of spares (2,016 lbs) and support an on-orbit maintenance period of over 56 hours. An MSM weight of 6,150 lbs was assumed.

Table IVD-20 Approach 2 - Intelsat Geosynchronous Mission Budgets

Event	Duration (hrs)	Inerts/* Losses (lbs)		APS (lbs)		MPS ΔV (ft/sec)	Initial Vehicle Weight (lbs)
		A	B	A	B		
First Shuttle/Tug Launch and Wait	48.0	144.0					121,231
Second Shuttle/Tug/MSM Launch	2.0	6.0					121,087
MSM Checkout	1.0	3.0	3.0				121,081
Tug Mating	1.0	3.0	3.0				121,075
Separation and Checkout	3.0	10.0	10.0	10.0			121,069
Phase in Shuttle Orbit (160 n mi)	11.0	46.0	46.0	30.0			121,039
Phasing/Plane Change (2°) Burn	0.35					6800	120,917
Coast 1 Rev. in Phasing Orbit	5.0	20.0	20.0	10.0	10.0		75,751
Tugs Separate							75,691
Tug A Deboost for Return	0.08					6800	11,010
Rendezvous and Docking	4.0			32.0			6,898
Tug A: Propellant Consumed 49,279 lbs							6,866
Tug A: Propellant Reserve 910							
Tug A: APS used 82							
Tug A: APS reserve 29							
Tug A: Tug adapter 200							
Tug B Inject into Geosynchronous Transfer (perigee burn and 2-1/2° plane change)	0.10					1489	64,681
Coast to Midcourse Correction	1.5		6.0		14.0		58,385
Midcourse Correction	0.03					50	58,365
Coast to 19,300 n mi Apogee	3.96		16.0		14.0		58,127
Circularize and Plane Change (24°) at Geosynchronous Altitude	0.12					5692	58,097
Rendezvous and Docking	6.0		24.0		97.0		39,280
On-Orbit Maintenance	56.0		224.0		40.0		39,159
Phase at Geosynch for Nodal Crossing	11.4		45.0		11.0		38,895
Deboost Burn	0.08					5840	38,839
Coast to Midcourse Correction	1.0		6.0		6.0		25,992
Midcourse Correction	0.01					15	25,978
Coast to 170 n mi Perigee	4.2		24.0		8.0		25,950
Inject into Return Phasing	0.05					3791	25,918
Coast 1 Rev. in Phasing Orbit	3.0		18.0		8.0		19,970
Circularize at 170 x 170 n mi orbit	0.05					4243	19,944
Rendezvous and Docking	4.0				32.0		14,896
TOTALS		232.0	451.0	82.0	242.0		

Tug B		
Tug B	Propellant Consumed	49,222
	Propellant Unused Capacity	943
	APS used	242
	APS unused capacity	75

Returned Weight		14,864
Tug dry weight	5,130	
Unusable Residuals	576	
APS reserve	29	
Propellant reserve	943	
Spares	2,016	
MSM**	6,150	
	14,864	

*Boil-off, fuel cell consumables.
**No life support system depletion - worst return case.

Again, all factors considered on this approach for the Intelsat are very similar to those for the DWS. The following capabilities are required because of this approach to maintain the Intelsat:

<u>Intelsat</u>

1) Capability to retract and deploy the solar arrays;
2) Receptacle for umbilical attachment from the MSM;
3) Laser radar reflectors (corner cubes);
4) Replaceable subsystem modules, including solar arrays;
5) Docking frame, probe, and latches compatible with the MSM;
6) Solar array booms capable of withstanding docking loads;
7) Redundant shutoffs, structural safety factors, no sharp edges or protrusions, and similar aspects to minimize hazards to EVA crewmen.

MSM

1) Docking provisions compatible with Intelsat;
2) Life support systems for IVA and EVA;
3) Umbilical system capable of conveying control commands and electrical power to the Intelsat;
4) Backup means of separation from the Intelsat in the event of failure of capture-latches to open;
5) Capability to control Tug APS (auxiliary propulsion) maneuvers;
6) Contingency life support and electrical power reserves.
7) Transmission of life support system status to ground through the Tug;
8) Attitude control capability to stabilize MSM for docking with rescue Tug;
9) Provisions for undocking/docking with the Tug in orbit. Includes laser reflectors;
10) Provisions for general flood lighting in the Intelsat system modules area and for portable lighting in other maintenance areas;
11) Provisions for transferring spares between the maintenance areas and the MSM hatch.

Tug

1) Compatibility with manned mission (man-rated);
2) Relay of data from the Intelsat/MSM to the ground;
3) Relay of communications between crewmen and ground;
4) Relay of data and communications between the MSM/Tug and the Orbiter during P/L bay checkouts;
5) Provisions for docking with MSM in orbit. Laser radar capability with or without MSM attached.

Orbiter

1) Provisions for attaching airlock transfer tunnel to MSM and retracting tunnel.

c. Approach 3 - Tables IVD-21 and IVD-22 present the mission budgets for the retrieval and delivery missions. From similarity to the DWS maintenance, no incompatibilities or problems are foreseen. The solar array mountings should withstand the Tug boost loads without folding the booms. However, the capability should exist to manually refold the array booms to a prelaunch configuration in the event it is desired to load the satellite in the P/L bay and return it for ground refurbishment.

Table IVD-21 Approach 3 - Intelsat Geosynchronous Retrieval Mission Budgets

Event	Duration Hours	Inerts/* Losses (lbs)	APS (lbs)	MPS ΔV (ft/sec)	Initial Vehicle Weight (lbs)
Tug Separation from Orbiter	2.0	10.0	8.6		56,625
Phase in Shuttle Orbit (160 n mi)	11.0	46.0	21.4		56,608
Phasing/Plane Change Burn	0.13			4494	56,541
Coast 1 Rev. in Phasing Orbit	3.0	5.0	17.5		41,510
Inject into Geosynchronous Transfer (Perigee Burn)	0.11			3672	41,487
Coast to Midcourse Correction	1.5	6.0	13.8		32,229
Midcourse Correction	0.03			50	32,209
Coast to 19,300 n mi Apogee	3.46	16.0	14.0		32,078
Circularize at Geosynchronous Altitude (Apogee Burn)	0.12			5828	32,048
Rendezvous and Docking	4.0	16.0	96.5		21,466
Phase in Orbit for Nodal Crossing	11.4	45.0	11.2		24,063
Deboost Burn	0.08			5840	24,007
Coast to Midcourse Correction	1.0	6.0	7.5		16,066
Midcourse Correction	0.02			35	16,053
Coast to 170 n mi Perigee	4.2	24.0	8.1		16,006
Inject into Return Phasing Orbit	0.05			3791	15,974
Coast 1 Rev. in Phasing Orbit	3.0	18.0	7.8		12,308
Circularize at 170 x 170 n mi Orbit	0.05			4243	12,282
Rendezvous and Dock with Orbiter	4.0		32.4		9,174
TOTALS	49.2	192.0	238.8		

Propellant Consumed	49,764	Returned Weight		9,141
Unused Propellant Capacity	396	Tug Dry Weight	5,150	
APS Consumed	239	Unusable Residuals	576	
Unused APS Capacity	78	APS Reserve	29	
		Propellant Reserve	396	
		Docking Frame	250	
		Intelsat	2,740	
			9,141	

*Boiloff, fuel cell consumables

Table IVD-22 Approach 3 - Intelsat Geosynchronous Delivery Mission Budgets

Event	Duration Hours	Inerts/* Losses (lbs)	APS (lbs)	MPS ΔV (ft/sec)	Initial Vehicle Weight (lbs)
Tug Separation from Orbiter	2.0	10.0	8.6		59,411
Phase in Shuttle Orbit (160 n mi)	11.0	46.0	21.4		59,392
Phasing/Plane Change Burn	0.13			4494	59,325
Coast 1 Rev. in Phasing Orbit	3.0	5.0	17.5		43,553
Inject into Geosynchronous Transfer (Perigee Burn)	0.11			3672	43,530
Coast to Midcourse Correction	1.5	6.0	13.8		33,816
Midcourse Correction	0.03			50	33,796
Coast to 19,300 n mi Apogee	3.46	16.0	14.0		33,658
Circularize at Geosynchronous Altitude (Apogee Burn)	0.12			5828	33,628
Coast and Orbit Trim	12.0	48.0	96.5		22,523
Deploy DMS	1.0	4.0	40.0		22,379
Phase in Orbit for Nodal Crossing	11.4	45.0	11.2		19,625
Deboost Burn	0.08			5040	19,569
Coast to Midcourse Correction	1.0	6.0	7.5		13,096
Midcourse Correction	0.02			35	13,083
Coast to 170 n mi Perigee	4.2	24.0	8.1		13,045
Inject into Return Phasing Orbit	0.05			3791	13,013
Coast 1 Rev. in Phasing Orbit	3.0	18.0	7.8		10,026
Circularize at 170 x 170 n mi	0.05			4243	10,000
Rendezvous and Dock with Orbiter	4.0		32.4		7,469
TOTALS	58.3	228.0	278.8		

Propellant Consumed	48,757	Returned Weight		7,437
Unused Propellant Capacity	1,432	Tug dry weight	5,150	
APS Consumed	279	Unusable residuals	576	
Unused APS Capacity	38	APS reserve	29	
		Propellant reserve	1,432	
		Docking Frame	250	
			7,437	

*Boiloff, fuel cell consumables

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The following additional capabilities are required because of this approach to maintenance of the Intelsat:

Intelsat

1) Capability to retract and deploy the solar arrays several times;
2) Receptacle for umbilical attachment from the Tug;
3) Laser radar reflectors (corner cubes);
4) Replaceable subsystem modules, including solar arrays;
5) Docking frame probe and latches compatible with the Tug;
6) Solar array booms capable of withstanding docking loads and Tug boost loads;
7) Redundant shutoffs, structural safety factors, no sharp edges or protrusions, and provisions for mounting portable foot restraints.

Tug

1) Relay of data between the Intelsat and the ground;
2) Relay of commands and power from the Shuttle to the Intelsat;
3) Provisions for docking with the Intelsat.

Shuttle Orbiter

1) Provide portable foot restraints;
2) Provide portable lights.

4. SEOS Maintenance

a. Approach 1 - As seen from previous calculations, only about 1200 lbs of spares (in addition to a 1150 lb servicer) could be transported to and from the SEOS, using a baseline Tug. This is about 67% of the total complement of replaceable units.

No particular problems are foreseen from docking with the SEOS. Depending on the type docking frame adopted, the solar array panels may need to be stowed prior to docking. The telescope doors should be closed by remote command prior to servicer rendezvous to minimize telescope contamination from the Tug/servicer.

Timelines, communications, power sources, and commands required for maintenance of the SEOS would be very similar to those for the DWS and Intelsat. No incompatibilities are foreseen. The following capabilities are required because of this approach to maintenance of the SEOS:

SEOS

1) Capability to retract and deploy the solar arrays;
2) Receptacle for umbilical attachment from the servicer;
3) Laser radar reflectors (corner cubes);
4) Replaceable subsystem modules, including solar arrays;
5) Docking frame and latches compatible with the servicer;
6) Capability, through remote commands, to close and open the telescope doors.

Servicer

1) Docking frame compatible with SEOS;
2) Servicing system controlled by commands from Tug;
3) Servicing system capable of preprogrammed or remote-control changeout of SEOS modules. The latter will incorporate the use of TV;
4) Umbilical system capable of being connected to the SEOS to convey control commands and electrical power;
5) Backup means of separation in the event capture latches fail to open.

Tug

1) Provide computer preprogrammed control of the servicer mechanisms;
2) Relay of remote control commands to the servicer;
3) Relay of data from the SEOS to the ground;
4) Relay of data from the servicer to the ground or to the orbiter during P/L bay checkouts.

b. Approach 2 - This approach, as previously determined, requires two tandem Tugs but has more-than-adequate capability to carry a full complement of SEOS spares (1,784 lbs) and support an on-orbit maintenance period of over 56 hours.

Again, all factors considered on this approach for the SEOS are very similar to those for the DWS and Intelsat. The following capabilities are required because of this approach to maintain the SEOS:

SEOS

1) Capability to retract and deploy the solar arrays;
2) Receptacle for umbilical attachment from the MSM;
3) Laser radar reflectors (corner cubes);

SEOS (Cont'd)

4) Replaceable subsystem modules, including solar arrays;
5) Docking frame, probe, and latches compatible with the MSM;
6) Redundant shutoffs, structural safety factors, no sharp edges or protrusions, and similar aspects to minimize hazards to EVA crewmen;
7) Capability, through remote commands, to close and open the telescope doors.

MSM

1) Docking provisions compatible with SEOS;
2) Life support systems for IVA and EVA;
3) Umbilical system capable of conveying control commands and electrical power to the SEOS;
4) Backup means of separation from the SEOS in the event of failure of capture-latches to open;
5) Capability to control Tug APS (auxiliary propulsion) maneuvers;
6) Contingency life support and electrical power reserves;
7) Transmission of life support system status to ground through the Tug;
8) Attitude control capability to stabilize MSM for docking with rescue Tug;
9) Provisions for undocking/docking with the Tug in orbit. Includes laser reflectors;
10) Provisions for general flood lighting in the SEOS system modules area and for portable lighting in other maintenance areas;
11) Provisions for transferring spares between the maintenance areas and the MSM hatch.

Tug

1) Compatibility with manned mission (man-rated);
2) Relay of data from the SEOS/MSM to the ground;
3) Relay of communications between crewmen and ground;
4) Relay of data and communications between the MSM/Tug and the Orbiter during P/L bay checkouts;
5) Provisions for docking with MSM in orbit. Laser radar capability with or without MSM attached.

Orbiter

1) Provisions for attaching airlock transfer tunnel to MSM and retracting tunnel.

c. Approach 3 - The baseline Tug is not capable of retrieving the 3697 lb SEOS from geosynchronous orbit to the Shuttle at 160 n.mi. It may be possible to raise the Shuttle to a slightly higher orbit to decrease the Tug requirements. However, Shuttle Orbiter maneuvering capabilities, as well as the Tug's, would be limited. Therefore, it should be assumed that two Tugs would be required for SEOS retrieval in this approach. One Tug is capable of delivering the serviced SEOS back into geosynchronous orbit.

For return to the orbiter for servicing, the solar array panels would be stowed in the original launch configuration. Also the telescope doors would again need to be closed by remote-command prior to Tug rendezvous. For checkout of the SEOS in Shuttle orbit, the SEOS would be deployed and separated from the orbiter prior to opening the telescope doors, to minimize telescope contamination. The doors would then be commanded closed, the SEOS retrieved, and attached to the loaded delivery Tug.

The following additional capabilities are required because of this approach to maintenance of the SEOS:

SEOS

1) Capability to retract and deploy the solar arrays and telescope doors several times;
2) Receptacle for umbilical attachment from the Tug;
3) Laser radar reflectors (corner cubes);
4) Replaceable subsystem modules, including solar arrays;
5) Docking frame probe and latches compatible with the Tug;
6) Redundant shutoffs, structural safety factors, no sharp edges or protrusions, and provisions for mounting portable foot restraints.

Tug

1) Relay of data between the SEOS and the ground;
2) Relay of commands and power from the Shuttle to the SEOS;
3) Provisions for docking with the SEOS.

Shuttle Orbiter

1) Provide portable foot restraints;
2) Provide portable lights.

5. TDRS Maintenance

Maintenance considerations for the TDRS are very similar to those for the DWS. Since the TDRS weight, as well as spares weights, are less than the DWS, all mission budgets would be less stringent for the TDRS. Other differences between maintenance considerations for the TDRS and the DWS are:

1. No need to roll in the TDRS solar array prior to Tug docking.
2. Refolding the antennas, if required for Approach 3, would be easier to accomplish on the TDRS.
3. Checkout of the reserviced TDRS would perhaps be simpler since the Tug could supply the electrical power (about 400 watts required).
4. The antennas (with the drive motor) would be replaceable on the TDRS.
5. Pointing requirements for checkout of the serviced TDRS in Approach 3 should be easier to attain.

6. EOGP Maintenance

a. Approach 1 - As seen from the mission budget calculations for the Intelsat, only about 1,200 lbs of spares (in addition to an 1,150 lb servicer) could be transported to and from the EOGP, using a baseline Tug. This would be about 28% of the total complement of replaceable units. Two Tugs in tandem-launch would be needed to transport a servicer with a greater spares weight.

Several considerations accrue for EOGP maintenance that did not appear with the other satellites. Cautions must be taken during rendezvous and docking to assure that the Tug/servicer does not impact the long VLF antennas. These antennas are 33 feet to the side and extend 44 feet to the rear and 22 feet to the front of the EOGP. Prior to closure and docking at the rear, the gimballed scan platform must be swung to a forward position. Prior to docking at the earth side of the satellite, the large 30' and 60' antennas would need to be refolded to minimize potential impacts from the Tug/servicer. Clearance between a docked Tug and these antennas, if deployed, would be about 2 feet.

The EOGP configuration also results in new requirements for a servicer, not like those for previous satellites studied. For the EOGP, the servicer must reach into the satellite interior and extract modules by a radial movement toward the satellite pointing centerline (see Figure IVD-8).

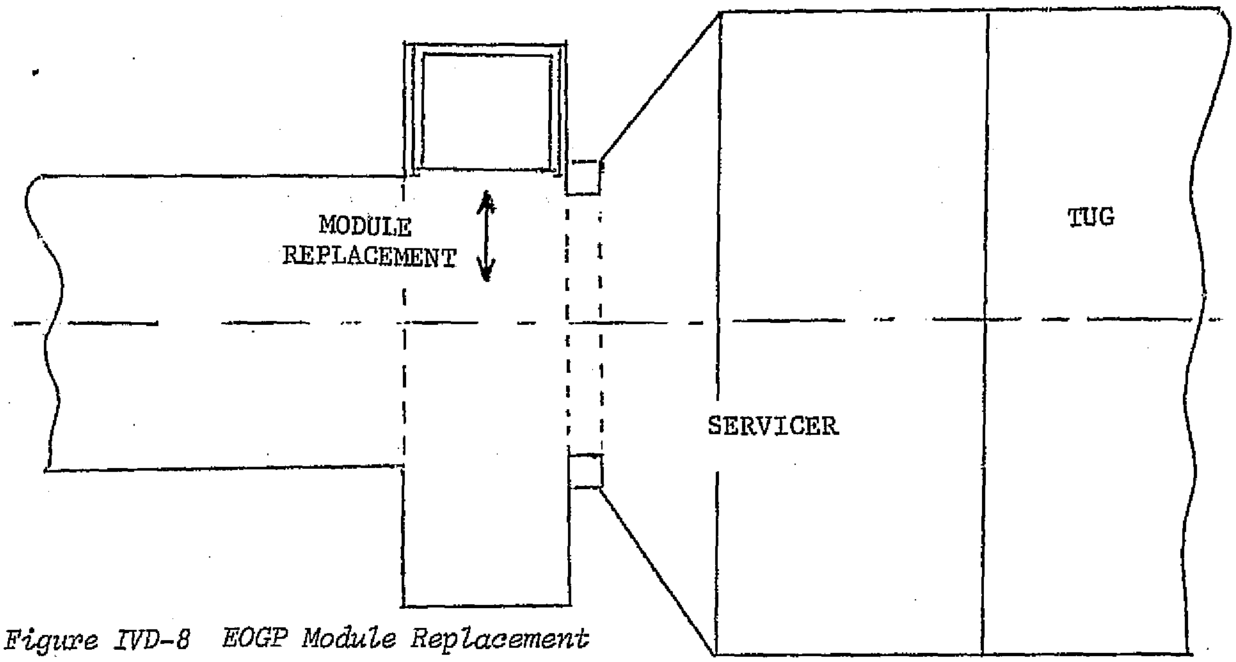

Figure IVD-8 EOGP Module Replacement

Some additional Tug fuel and time will be required for maneuvers to move the servicer from one EOGP docking port to the other. Also, because of the greater number of modules on the EOGP, more module replacements might be required on a maintenance mission to the EOGP. This would tend to make the maintenance period longer than for other satellites. Otherwise, timelines, communications, power sources, and commands required for maintenance of the EOGP would be very similar to those for the DWS and Intelsat. No incompatibilities are foreseen. The following capabilities are required because of this approach to maintenance of the EOGP:

<u>EOGP</u>

1) Capability to retract and deploy the solar arrays and the large antennas;
2) Receptacle for umbilical attachment from the servicer;
3) Laser radar reflectors (corner cubes);
4) Replaceable subsystem modules, including solar arrays;
5) Docking frame and latches at both ends, compatible with the servicer;
6) Capability to remotely swing the gimballed scan platform to a forward location.

<u>Servicer</u>

1) Docking frame compatible with the EOGP;
2) Servicing system controlled by commands from Tug;
3) Servicing system capable of preprogrammed changeout of EOGP modules or remote-control changeout of the replaceable units. The latter will incorporate the use of TV;

Servicer (Cont'd)

4) Umbilical system capable of being connected to the EOGP to convey control commands and electrical power;
5) Backup means of separation in the event capture latches fail to open.

Tug

1) Provide computer preprogrammed control of the servicer mechanisms;
2) Relay of remote control commands to the servicer;
3) Relay of data from the EOGP to the ground;
4) Relay of data from the servicer to the ground or to the orbiter during P/L bay checkouts.

b. Approach 2 - As seen from the mission budgets presented in Table IVD-23, only about 2850 lbs of spares could be carried to and from geosynchronous orbit (assuming tandem Tugs and a 6,150 lb MSM). This is about 66% of the total replaceable-units mass. The calculations indicate a Shuttle payload weight, for the second launch, of greater than 65,000 lbs. A better determination of the MSM weight would be needed in this case to verify whether the spares weight would be more limited.

Table IVD-23 Approach 2 - EOGP Geosynchronous Mission Budgets

Event	Duration (hrs)	Inerts/ ** Losses (lbs) A	Inerts/ ** Losses (lbs) B	APS (lbs) A	APS (lbs) B	MPS ΔV (ft/sec)	Initial Vehicle Weight (lbs)
First Shuttle/Tug Launch and Wait	48.0	144.0					122,095
Second Shuttle/Tug/MSM Launch*	2.0	6.0					121,951
MSM Checkout	1.0	3.0	3.0				121,945
Tug Mating	1.0	3.0	3.0				121,939
Separation and Checkout	3.0	10.0	10.0	10.0			121,933
Phase in Shuttle Orbit (160 n mi)	11.0	46.0	46.0	30.0			121,903
Phasing/Plane Change (2°) Burn	0.35					6800	121,781
Coast 1 Rev. in Phasing Orbit	5.0	20.0	20.0	10.0	10.0		76,293
Tugs Separate							76,233
Tug A Deboost for Return	0.08					6800	10,688
Rendezvous and Docking	4.0			32.0			6,696
							6,664
Tug A: Propellant Consumed 49,480 lbs							
Tug A: Propellant Reserve 709							
Tug A: APS used 82							
Tug A: APS reserve 29							
Tug A: Tug adapter 200							
Tug B Inject into Geosynchronous Transfer (perigee burn and 2-1/2° plane change)	0.10					1489	65,545
Coast to Midcourse Correction	1.5		6.0		14.0		59,165
Midcourse Correction	0.03					50	59,145
Coast to 19,300 n mi Apogee	3.96		16.0		14.0		58,904
Circularize and Plane Change (24°) at Geosynchronous Altitude	0.12					5692	58,874
Rendezvous and Docking	6.0		24.0		97.0		39,803
On-Orbit Maintenance	56.0		224.0		40.0		39,682
Phase at Geosynch for Nodal Crossing	11.4		45.0		11.0		39,418
Deboost Burn	0.08					5840	39,362
Coast to Midcourse Correction	1.0		6.0		8.0		26,342
Midcourse Correction	0.01					13	26,328
Coast to 170 n mi Perigee	4.2		24.0		8.0		26,299
Inject into Return Phasing	0.05					3791	26,267
Coast 1 Rev. in Phasing Orbit	3.0		18.0		8.0		20,239
Circularize at 170 x 170 n mi orbit	0.05					4243	20,213
Rendezvous and Docking	4.0				32.0		15,097
TOTALS		232.0	451.0	82.0	242.0		

Tug B		
	Propellant Consumed	49,885
	Propellant Unused Capacity	304
	APS used	242
	APS unused capacity	75

Returned Weight		15,065
Tug dry weight	5,150	
Unusable Residuals	576	
APS reserve	29	
Propellant reserve	310	
Spares	2,850	
MSM ***	6,150	
	15,065	

*Initial Shuttle p/L weight 65,637 lbs

**Boil-off, fuel cell consumables.

***No life support system depletion - worst return case.

Special provisions are needed during the EVA portion of the mission to minimize contamination of optical surfaces. These surfaces include the earth-pointing sensors in the forward equipment ring, star trackers in the common support ring, mirrors inside the 1.5-meter telescope, the removable rotating mirror at the rear of the telescope, and the receiving lens in the applicable modules in the aft equipment ring.

Effluent from the EVA crewman would be the dominating source of contamination. Some methods of minimizing contamination would be:

1) Completely closed-loop pressure suit;
2) Umbilical life support system recirculating all consumables to the MSM;
3) Use of a ducting system to collect and carry all effluents away from the work areas.

In addition to effluents, consideration should be given to suit design or pressure suit coveralls to minimize abrasion particles.

Another possible method to prevent optical contamination would be covers for the equipment. This would appear to be a less efficient method because of the number of optical surfaces and the need to position the covers remotely. For IVA shirtsleeve operations however, if this maintenance method were used, the remote-operated-covers method might be necessary. Use of clean-room type clothing would help, but might not alone be sufficient.

Again, all factors considered on this approach for the EOGP are very similar to those for the DWS. Timelines may vary somewhat during the maintenance period since in-orbit maneuvers will be made to change the MSM to the other EOGP docking port. Maximum quantity of modules to be replaced is unknown since the spares weight is limited.

The following capabilities are required because of this approach to maintain the EOGP:

<u>EOGP</u>

1) Capability to retract and deploy the solar arrays and large antennas;
2) Capability to swing the scan platform to a forward position by remote commands;
3) Receptacle for umbilical attachment from the MSM;

EOGP (Cont'd)

4) Laser radar reflectors (corner cubes);
5) Replaceable subsystem modules, including solar arrays and other external units;
6) Docking frame, probe, and latches compatible with the MSM;
7) Redundant shutoffs, structural safety factors, no sharp edges or protrusions, and similar aspects to minimize hazards to EVA crewmen;
8) Provisions for remote-controlled optics covers should IVA shirtsleeve maintenance capabilities be instigated.

EVA Support System

1) Pressure suit and/or life support system methods for minimizing effluents that would contaminate optical surfaces.

MSM

1) Docking provisions compatible with EOGP;
2) Life support systems for IVA and EVA;
3) Umbilical system capable of conveying control commands and electrical power to the EOGP;
4) Backup means of separation from the EOGP in the event of failure of capture-latches to open;
5) Capability to control Tug APS (auxiliary propulsion) maneuvers;
6) Contingency life support and electrical power reserves;
7) Transmission of life support system status to ground through the Tug;
8) Attitude control capability to stabilize MSM for docking with rescue Tug;
9) Provisions for undocking/docking with the Tug in orbit. Includes laser reflectors;
10) Provisions for general flood lighting in the EOGP system modules area and for portable lighting in other maintenance areas;
11) Provisions for transferring spares between the maintenance areas and the MSM hatch.

Tug

1) Compatibility with manned mission (man-rated);
2) Relay of data from the EOGP/MSM to the ground;
3) Relay of communications between crewmen and ground;
4) Relay of data and communications between the MSM/Tug and the Orbiter during P/L bay checkouts;
5) Provisions for docking with MSM in orbit. Laser radar capability with or without MSM attached.

Orbiter

1) Provisions for attaching airlock transfer tunnel to MSM and retracting tunnel.

c. Approach 3 - Single baseline Tugs are not capable of retrieving or deploying the EOGP. Therefore, it would require tandem Tugs to retrieve the EOGP from geosynchronous orbit and it would require tandem Tugs to deliver the serviced EOGP back to geosynchronous orbit. This requires four Shuttle/Tug flights for a single maintenance mission. Although this approach appears impractical on this basis, it is possible that the use of a SEPS for inter-orbit transfers might make the approach more attractive.

For return from geosynchronous orbit, all EOGP appendages (solar arrays, antennas, and sensor booms) must be retracted to the initial launch configuration.

Contamination of EOGP optical surfaces could be a problem during servicing at the orbiter, depending on the extent of a contaminant cloud about the orbiter. Optics covers, as discussed in Approach 2, may be required.

The following additional capabilities are required because of this approach to maintenance of the DWS:

EOGP

1) Capability of multiple retractions and deployments of the EOGP appendages;
2) Receptacle for umbilical attachment from the Tug;
3) Laser radar reflectors (corner cubes);
4) Replaceable subsystem modules, including solar arrays, antennas, and gimballed scan platform;
5) Docking frame probe and latches compatible with the Tug;
6) Redundant shutoffs, structural safety factors, no sharp edges or protrusions, and provisions for mounting portable foot restraints,
7) Remote controlled optics covers.

Tug

1) Relay of data between the EOGP and the ground;
2) Relay of commands and power from the Shuttle to the EOGP;
3) Provisions for docking with the EOGP.

Shuttle Orbiter

1) Provide portable foot restraints;
2) Provide portable lights.

7. Radio Astronomy Telescope Maintenance

Maintenance of the Radio Astronomy Telescope will be investigated assuming the three approaches considered for the other satellites. However, because of configuration and orbit differences from the other satellites, some variations to these approaches are needed.

a. Approach 1 - In this approach, an Earth Orbital Teleoperator System (EOTS) attached inside the Tug docking structure (see Figure IVD-9) will be boosted to the telescope orbit. Once docked, the EOTS will act as the servicer and through ground control will exchange the telescope modules with spares carried on the Tug. The Tug would then separate and loiter near the telescope. The EOTS would then be deployed from the Tug and flown to those telescope ACS and star tracker modules needing to be replaced. The EOTS would dock on the rib structure (see Figure IVC-18). The teleoperator would replace the modules with spares carried on the EOTS.

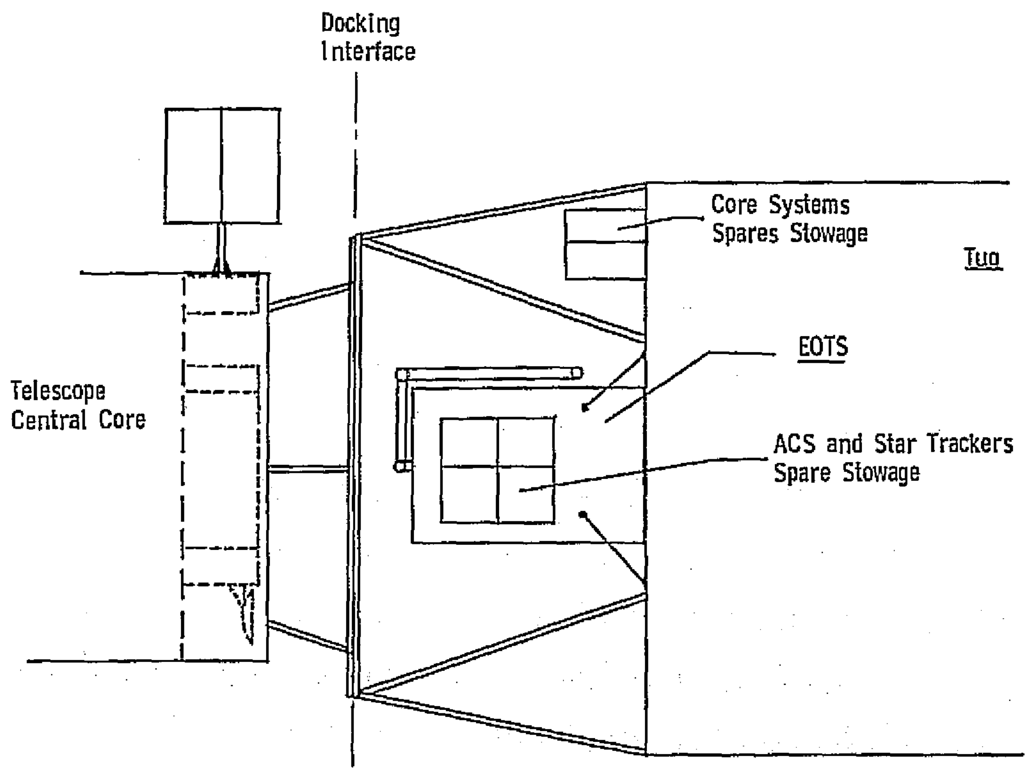

Figure IVD-9 TUG/EOTS Servicing Interfaces with RAT Central Core

This potential need for a servicer that must move to various locations on large-structure space systems leads to consideration of the EOTS as the servicer for all maintenance missions. This concept is discussed further in IV.D.10.

Based on previous calculations, it is obvious that the baseline Tug is capable of transporting a 500 to 1,000-lb EOTS, 830 lbs of spares, and associated docking hardware to the 8,000 n mi telescope orbit and return. Based on similarities to previous satellites studied, no incompatibilities or problems are foreseen with this approach.

The following capabilities are required because of this approach to maintenance of the radio astronomy telescope:

Radio Astronomy Telescope

1) Capability to retract and deploy the solar arrays;
2) Receptacle for umbilical attachment from the Tug;
3) Laser radar reflectors (corner cubes);
4) Replaceable subsystem modules, including solar arrays;
5) Docking frame and latches compatible with the Tug.

EOTS Servicer

1) Docking provisions compatible with telescope ribs docking points;
2) Teleoperator controlled by commands through Tug while attached to Tug;
3) Backup means of separation in the event docking latches fail to open;
4) Provide stowage for spare modules.

Tug

1) Umbilical system capable of being connected to the telescope to convey control commands and electrical power;
2) Relay of remote control commands to the EOTS;
3) Relay of data from the telescope to the ground;
4) Relay of data from the EOTS to the ground or to the orbiter during P/L bay checkouts;
5) Provide docking points for the EOTS inside the telescope docking frame;
6) Provide stowage for spare modules.

b. <u>Approach 2</u> - Because of an orbit of only 8,000 n miles, a single baseline Tug is capable of boosting a 6,150 lb MSM and 830 lbs of spares to the radio astronomy telescope and returning. Table IVD-24 presents the mission budgets. Either EVA crewmen (with manned maneuvering units (MMU) or EOTS's will be used to service the external replaceable modules (star trackers and ACS pods). The unused propellant reserve capacity shown in Table IVD-24 indicates this extra hardware weight could be carried.

Table IVD-24 Approach 2 - RAT Geosynchronous Mission Budgets

Event	Duration (hrs)	Inerts/* Losses (lbs)	APS (lbs)	MPS ΔV (ft/sec)	Initial Vehicle Weight (lbs)
Shuttle Ascent and Tug Preseparation	8.0	24.0			63,375
Tug Separation from Orbiter	2.0	10.0	8.6		63,351
Phase in Shuttle Orbit (160 n mi)	11.0	46.0	21.4		63,332
Phasing/Plane Change Burn	0.10			3,400	63,265
Coast 1 Rev. in Phasing Orbit	2.0	4.0	15.0		50,074
Inject into Geosynchronous Transfer (Perigee Burn)	0.09			2,498	50,055
Coast to Midcourse Correction	1.0	4.0	12.0		42,154
Midcourse Correction	0.03			50	42,138
Coast to 8,000 n mi Apogee	1.5	6.0	5.0		41,966
Circularize at Geosynchronous Altitude (Apogee Burn)	0.11			4,371	41,955
Rendezvous and Docking	4.0	15.0	96.5		31,062
On-Orbit Maintenance and Checkout	13.5	54.0	8.0		30,951
Phase at Geosynch for Nodal Crossing	11.4	45.0	11.2		30,889
Deboost Burn	0.06			4,385	30,833
Coast to Midcourse Correction	1.0	6.0	7.5		22,806
Midcourse Correction	0.01			13	22,792
Coast to 170 n mi Perigee	1.0	4.0	5.0		22,768
Inject into Return Phasing	0.04			2,619	22,759
Coast 1 Rev. in Phasing Orbit	2.0	4.0	6.0		19,008
Circularize at 170 x 170 n mi Orbit	0.05			3,150	18,998
Rendezvous and Dock with Orbiter	4.0		32.4		15,297
TOTALS	60.9	222.0	228.6		

Propellant Consumed	47,660
Unused Propellant Capacity	2,529
APS Consumed	229
Unused APS Capacity	88

<u>Returned Weight</u>		15,265
Tug dry weight	5,150	
Unusable residuals	576	
APS reserve	29	
Propellant reserve	2,530	
Spares	830	
MSM**	6,150	
	15,265	

* Boil off, fuel cell consumables.
**No life support system depletion assumed

The following capabilities are required because of this approach:

Radio Astronomy Telescope

1) Capability to retract and deploy the solar arrays;
2) Receptacle for umbilical attachment from the MSM;
3) Laser radar reflectors (corner cubes);
4) Replaceable subsystem modules, including solar arrays;
5) Docking frame, probe, and latches compatible with the MSM;
6) Redundant shutoffs, structural safety factors, no sharp edges or protrusions, and similar aspects to minimize hazards to EVA crewmen.

MSM

1) Docking provisions compatible with radio astronomy telescope;
2) Life support systems for IVA and EVA;
3) Umbilical system capable of conveying control commands and electrical power to the telescope;
4) Backup means of separation in the event of failure of capture-latches to open;
5) Capability to control Tug APS (auxiliary propulsion) maneuvers;
6) Contingency life support and electrical power reserves;
7) Transmission of life support system status to ground through the Tug;
8) Attitude control capability to stabilize MSM for docking with rescue Tug;
9) Provisions for undocking/docking with the Tug in orbit. Include laser reflectors;
10) Provisions for general flood lighting in the system modules area and for portable lighting in other maintenance areas;
11) Provisions for transferring spares between the maintenance areas and the MSM hatch;
12) Provide EOTS or MMU translation capabilities for servicing external modules.

Tug

1) Compatibility with manned mission (man-rated);
2) Relay of data from the telescope/MSM to the ground;
3) Relay of communications between crewmen and ground;
4) Relay of data and communications between the MSM/Tug and the Orbiter during P/L bay checkouts;
5) Provisions for docking with MSM in orbit. Laser radar capability with or without MSM attached.

Orbiter

1) Provisions for attaching airlock transfer tunnel to MSM and retracting tunnel.

c. Approach 2 - A single baseline Tug is not capable of retrieving the radio astronomy telescope from the 8,000 n mi orbit. This would require two Tugs. A single Tug could deliver the telescope back to the 8,000 n mi orbit, however.

Other problems make this approach undesirable for the radio astronomy telescope. Depending on the telescope structure and assembly methods, Tug transportation may not be acceptable.

The following additional capabilities are required because of this approach:

Radio Astronomy Telescope

1) Capability to retract and deploy the telescope beams/net and the solar arrays several times;
2) Receptacle for umbilical attachment from the Tug;
3) Laser radar reflectors (corner cubes);
4) Replaceable subsystem modules, including solar arrays;
5) Docking frame probe and latches compatible with the Tug;
6) Redundant shutoffs, structural safety factors, no sharp edges or protrusions, and provisions for mounting portable foot restraints.

Tug

1) Relay of data between the telescope and the ground;
2) Relay of commands and power from the Shuttle to the telescope;
3) Provisions for docking with the telescope.

Shuttle Orbiter

1) Provide portable foot restraints.
2) Provide portable lights.

8. Maintenance Approach Summary

The most significant results derived from the analyses of the three maintenance approaches are summarized in Table IVD-25.

Table IVD-25 Significant Results of Maintenance Approach Analyses

SATELLITE	APPROACH 1 SERVICER IN HEO	APPROACH 2 MSM IN HEO	APPROACH 3 RMS/EVA IN LEO
DWS	Solar arrays must be retracted. Near Tug capability for full spares complement.	Two Tugs required. Sufficient capability for full spares complement. Solar arrays must be retracted.	Solar array must be retracted. Antenna probably must be refolded. Two Tugs required. LEO pointing and checkout of antenna more difficult and time consuming.
TDRS	No need to retract solar array except for replacement. Sufficient capability for full spares complement.	Two Tugs required. Sufficient capability for full spares complement.	Antennas easier to refold. Two Tugs required.
INTELSAT	Solar arrays must be retracted. Tug capability for 60% spares replacement.	Two Tugs required. Sufficient capability for full spares complement. Solar arrays must be retracted.	Solar arrays must be retracted. Two Tugs required.
SEOS	Solar arrays must be retracted. Tug capability for 67% spares replacement.	Two Tugs required. Sufficient capability for full spares complement. Solar arrays must be retracted. Optical contamination protection required.	Three Tugs required. Solar arrays must be retracted. Optical contamination protection required.
EOGP	Tug capability for 28% spares replacement. Solar arrays must be retracted. Antennas and scanner platform must be retracted. VLF antenna creates caution during rendezvous. Docking at both ends of satellite. Longer timeline because of more modules.	Two Tugs required. Tug capability for 66% spares replacement. Optical contamination protection required. Solar arrays must be retracted. Antennas and scanner platform must be retracted. VLF antenna creates caution during rendezvous. Docking at both ends of satellite. Longer timeline because of more modules.	Four Tugs required. All appendages must be retracted and again deployed. Optical contamination protection required.
RADIO ASTRONOMY TELESCOPE	Sufficient Tug capability for full spares complement. Free-flyer type servicer required for star trackers and propulsion modules.	Single Tug required. Sufficient capability for full spares complement. Free-flyer or MMU required for replacing star trackers and propulsion modules.	Three Tugs required. Possibly could not return satellite - depending on configuration.

These analyses also lead to the following general requirements for maintenance of satellites.

General Requirements for Servicer

1) Docking provisions compatible with the satellite and integrated with the Tug rendezvous and docking systems (if docking is between the satellite and the servicer);
2) Servicing system controlled by instructions from preprogrammed Tug computer circuitry or by commands from ground sources;
3) Lighting and TV aids for remote control module changeout;
4) Umbilical system for docking engagement to satellite to convey control commands and electrical power;
5) Backup means of separation in the event of docking latch failure to open;
6) Provide stowage provisions for replaceable spares;
7) Servicing system capable of reaching and exchanging all replaceable units on the subject satellite;
8) Servicer end-effector compatible with the satellite module latch mechanisms.

General Requirements for Satellites

1) Capability to retract appendages (solar arrays, antennas, external experiments, etc.) that are not able to withstand docking impact loads or that may impact the docking system (reasonable maneuvering space required);
2) Capability to command retraction of appendages (item 1) by signals from remote sources (ground, orbiter, TDRS);
3) Capability to deploy appendages by remote command and hardline link through the servicing system;
4) Capability for multiple deployment and retraction of appendages for Approach 3 maintenance;
5) Laser radar reflectors (docking aids) and other docking provisions compatible with servicing system;
6) Receptacle for umbilical attachment from servicing system;
7) Circuitry for disengaging selected satellite equipment functions by remote control or through the umbilical from the servicing system;
8) All functional systems (excluding such equipment as passive antennas) replaceable as modules or self-contained units;
9) Module latch mechanisms should be compatible with capabilities of servicer end-effectors or hand-held EVA tools;
10) For EVA maintenance, redundant fluid and mechanical shutoffs, structural safety factors, and elimination of sharp edges and protrusions are required to minimize hazards to EVA crewmen;

General Requirements for Satellites (Cont'd)

11) Capability of remotely commanding opening and closing of covers on contamination sensitive optical equipment.

General Requirements for Shuttle/Tug

1) Provisions for Tug docking directly with the satellite in Approach 3 and also in Approach 1 if the servicer is separate equipment installed inside the Tug docking frame;
2) Tug computer and circuitry to provide preprogrammed instructions to the servicer (if applicable);
3) Tug circuitry to relay remote commands and power to the servicer and/or satellite;
4) Tug relay of data from the servicing system and/or satellite to the ground or to the orbiter during checkouts;
5) Backup means of separation in the event of docking latch failure;
6) Provide external stowage provisions for large replaceable units such as solar array and antenna packages;
7) Provisions for P/L bay stowage (including environmental protection) of replaceable units for Approach 3;
8) Provide portable foot restraints and lighting in Approach 3;
9) Adapter for tandem Tug operations.

9. Maintenance Mission Shuttle Transportation System Requirements

This section develops the requirements for specific STS flights to support the three maintenance approaches. Since the Traffic Model only schedules the DWS, Intelsat, SEOS, and TDRS, only these satellites will be considered in further maintenance approach comparisons and tradeoffs (Section IV.F).

Table IVD-26 presents the satellite schedules based on the Traffic Model and SSPD. This schedule has many inconsistencies between operations and the satellite mean mission durations (MMD). This schedule was revised in the Integrated Orbital Servicing Study (IOSS)[1] to be compatible with servicing plans and to be more consistent (Table IVD-27). Figure IVD-10 presents the locations of the satellites in orbit.

Table IVD-28 summarizes the satellite replaceable units and weights based on the reconfigured servicable versions of these satellites, as derived in this study.

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[1] Integrated Orbital Servicing Study for Low-Cost Payload Programs, Contract NAS8-30820, Martin Marietta Corporation for Marshall Space Flight Center.

Table IVD-26 Traffic Model/SSPD Schedules

	MMD (YRS)		1981	1982	1983	1984	1985	1986	1987	1988	1989	1990	1991
DWS	5	New Launch	1	1			1						
		Refurb										1	
		Retrieve						1	1				
INTELSAT	10	New Launch			2	3	2	2					
		Refurb									2	3	2
		Retrieve							1	1	2	3	
TDRS	5 (7)*	New Launch			3					3			
		Refurb											
		Retrieve											
SEOS	8/5 (2)*	New Launch	1		1		1		2		2		2
		Refurb											
		Retrieve						1					

*SSPD data

Table IVD-27 IOSS Modified Traffic Model (Serviceable Satellites)

	AOT* (YEARS)		81	82	83	84	85	86	87	88	89	90	91
DWS	5	Del		2									
		Serv							2				
INTELSAT	6	Del			2	3	2	2					
		Serv									2	3	2
TDRS	5	Del			3								
		Serv								3			
SEOS	2	Del			1				1				
		Serv					1		1		2		2

*AOT - Average operational time

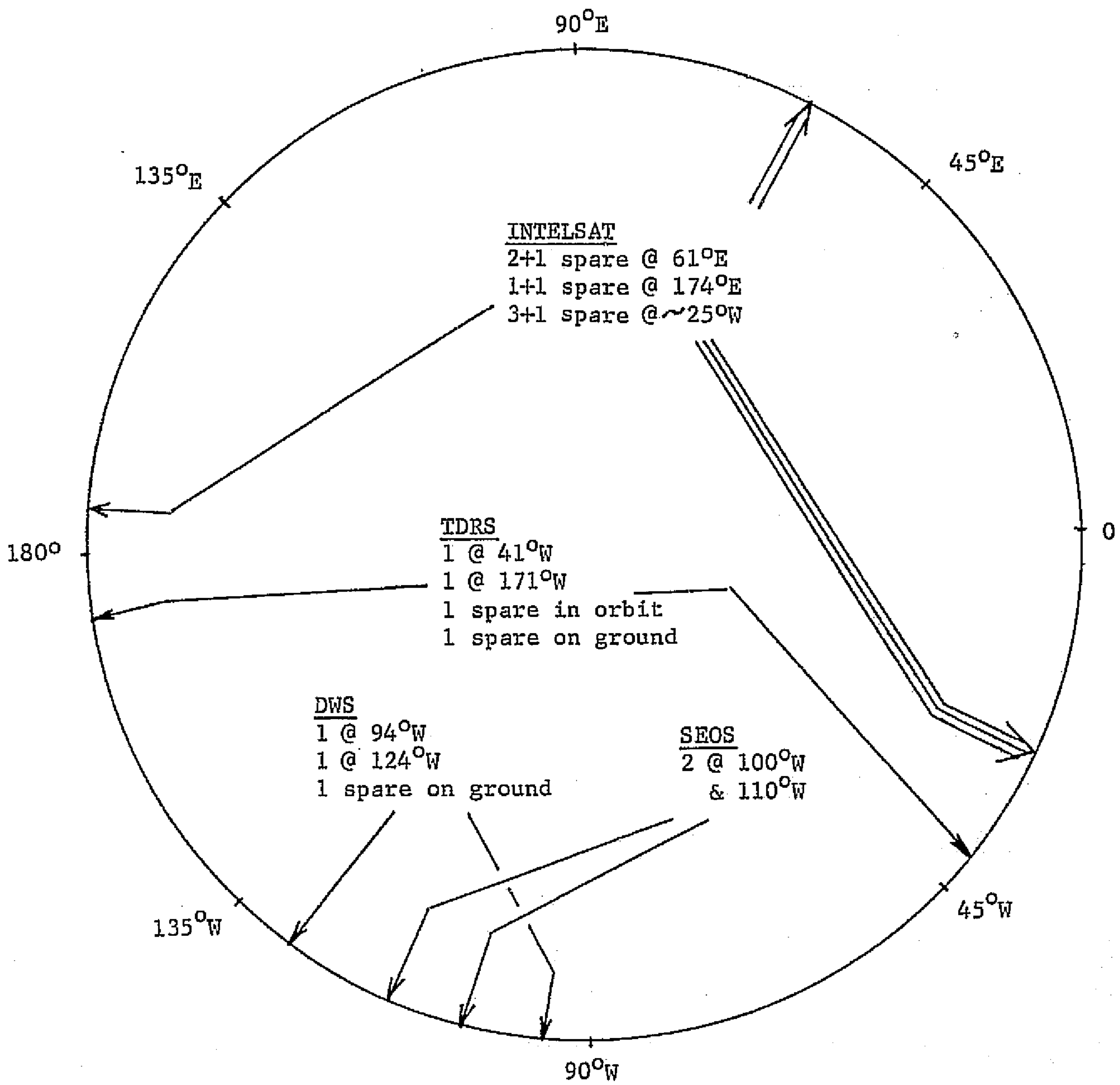

Figure IVD-10 Orbital Positions

Table IVD-28 Replaceable Units Weight Summary

	REPLACEABLE UNITS (QUANTITY)/WEIGHT, LBS EACH																				
SATELLITE	Solar Array Assy	Attitude Control #1	Attitude Control #2	ACS Propulsion	Power Module #1	Power Module #2	TT&C	Data Processing	LDR Transmitter & TT&C #1	LDR Transmitter and TT&C #2	HDR/MDR	TDRS/GS	Electronics Mod.	HDR/MDR Ant.	TDRS/GS Ant.	Receivers	Mission Equip.	Transponders	Power Monitor	Filters/S-Band	Amplifiers	Total
DHS	(2) 263 (254 dry)	(1) 130		(2) 100 (45 dry)	(1) 179		(2) 27												(2) 20	(2) 10	(2) 90	(14) 1329 (1201 dry)
INTELSAT	(2) 135	(2) 75		(4) 135 (69 dry)	(2) 65	(2) 75	(2) 40									(2) 48		(8) 75				(24) 2016 (1752 dry)
SEOS	(2) 112	(2) 206		(2) 171 (77 dry)	(2) 126		(1) 85	(1) 61									(2) 204					(12) 1784 (1596 dry)
TDRS	(1) 77	(1) 38	(1) 32	(2) 56 (36 dry)	(1) 59	(1) 54			(1) 37	(1) 31	(2) 44	(1) 46	(1) 41	(2) 33	(1) 14					(1) 51		(17) 746 (706 dry)

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Maintenance approaches will be analyzed in a manner similar to that used in the Integrated Orbital Servicing Study (IOSS). That is, the satellites are assumed serviced at the end of each average operational time (AOT) period. The mass of modules replaced is determined by calculating a parts factor (PF), which is based on a combination of random failures and wearout. The calculations of PF in the IOSS did not include the solar panels as replaceable units. Therefore, and since these are considered replaceable in this study, solar panels will be added to the PF calculations. The PF factor is multiplied times the satellite weight to determine the module replaced weights at the end of the AOT period. This . cheme, when applied over a long period, is comparable, on the average, to the replacement schedules calculated later in the geosynchronous maintenance vehicle analysis (Section IV.E), based on failure rates.

Table IVD-29 presents the predicted module replacement and return weights for each satellite. Returned propulsion modules are assumed empty of propellant.

Table IVD-29 Predicted Module Replacement Weights

	PF	SATELLITE WEIGHT (lbs)	SOLAR ARRAYS (lbs)	REPLACEMENT MODULES (lbs)	PROPELLANT WEIGHT (lbs)	RETURNED MODULES (lbs)
DWS	.28	1904	622	1155	128	1027
INTELSAT	.31	2740	270	1120	264	856
TDRS	.38	1139	77	510	39	471
SEOS	.19	3697	224	926	188	738

Replaced modules could be returned to the ground for refurbishment and subsequent use as spares. Spares costs would be reduced but STS penalties would be paid. However, module design might completely change over the program lifetime, making the refurbished spares obsolete. To discard replaced modules in orbit could create excessive space litter. Both of these modes will be investigated for costs in Section IV.F.

Table IVD-30 presents a breakdown of the missions required for the Tug/servicer maintenance approach. The number of STS flights is governed by the Tug payload delivery and return weights. The weights given include an 1150 lb servicer. Tug in-orbit transfers between satellites is limited to one per mission because of cryogenic boil-off, fuel cell consumables, and APS usage for rendezvous. Longitude phasing, between most of the subject satellites, will require about 2 or 3 days. Faster times would require excessive fuel consumption (high ΔV requirement).

Table IVD-30 TUG/Servicer Maintenance Missions

	YEAR	RETURN MODULES			EXPENDABLE MODULES		
	(Number of Satellites)	Weight Up (lbs)	Weight Down (lbs)	Number STS Flights	Weight Up (lbs)	Weight Down (lbs)	Number STS Flights
DWS	1987 (2)	2305 each	2177 each	2	3460 total	1150	1
INTELSAT	1989 (2) 1990 (3) 1991 (2) 1992 (2)*	2270 each	2006 each	2 3 2 2	3390 total {3390 (2) {2270 (1) 3390 total 3390 total	1150 1150 1150 1150 1150	1 1 1 1 1
TDRS	1988 (3)	2170 (2) 1660 (1)	2092 (2) 1621 (1)	1 1	2680 total	1150	1
SEOS	1985 (1) 1987 (1) 1989 (2) 1991 (2)	2076 2076 2076 each 2076 each	1888 1888 1888 each 1888 each	1 1 2 2	2076 2076 3002 (2) 3002 (2)	1150 1150 1150 1150	1 1 1 1
TOTAL STS FLIGHTS				19			11

*1992 included to put the analysis on a more common base with the on-orbit geosynchronous maintenance vehicle analysis.

With the manned servicing module (MSM) maintenance approach, Tandem Tugs are required because of the 6150 lb MSM. However, the Tugs have greater boost capability than single-satellite servicing. Tandem Tugs can boost the MSM and spares for servicing at least two satellites on each mission. Only one in-orbit transfer is permitted, however, because of time penalties, as discussed before. Table IVD-31 presents a breakdown on the MSM servicing missions. All three TDRS's can be serviced (as with the Tug/servicer) because two of these are at one location (see Figure IVD-10). Considerations for returning or discarding replaced modules have no effect on the number of required STS flights. However, these two options will be costed because of some potential cost savings from refurbishing returned modules.

Table IVD-31 MSM Maintenance Missions

	Year (Number of Satellites)	Weight Up (lbs)	Weight Down (lbs)	Number of STS Flights (Tandem Tugs)
DWS	1987 (2)	8460 (2)	8204 (2)	2
INTELSAT	1989 (2)	8390 (2)	7862 (2)	2
	1990 (3)	8390 (2)	7862 (2)	2
		7270 (1)	7006 (1)	2
	1991 (2)	8390 (2)	7862 (2)	2
	1992 (2)	8390 (2)	7862 (2)	2
TDRS	1988 (3)	7680 (3)	7563 (3)	2
SEOS	1985 (1)	7076 (1)	6888 (1)	2
	1987 (1)	7076 (1)	6888 (1)	2
	1989 (2)	8002 (2)	7626 (2)	2
	1991 (2)	8002 (2)	7626 (2)	2
TOTAL STS FLIGHTS				22

As an option to Approach 3 (retrieval for maintenance at the Shuttle orbiter), the mode of returning the satellite to the ground for refurbishment will also be considered.

Table IVD-32 presents the breakdown of mission schedules based on Tug capability to retrieve one 3400 lb satellite or deliver two 2600 lb satellites (approximately) and more without longitude separation. Retrieving more than one satellite on one Tug flight is not considered. In the case of maintenance at the orbiter, it is assumed that a short downtime is permissible for the DWS and TDRS and, therefore, maintenance on multiple satellites is performed simultaneously. This permits redelivery with one Tug each. For ground refurbishment, separate missions are assumed necessary.

Table IVD-32 Satellite Retrieval Maintenance Missions

	Year (Number of Satellites)	Satellite Weight - Each (lbs)	Tug Flights Required	
			Retrieve	Deliver
DWS	1987 (2)	1904	2	1 2*
INTELSAT	1989 (2) 1990 (3) 1991 (2) 1992 (2)	2740 2740 2740 2740	2 3 2 2	2 3 2 2
TDRS	1988 (3)	1139	3	1 3*
SEOS	1985 (1) 1987 (1) 1989 (2) 1991 (2)	3697 3697 3697 3697	2 2 4 4	1 1 2 2
TOTAL STS FLIGHTS				43 46*

*Separate missions required for ground refurbishment because of the long downtime.

10. Servicer Concept

The analysis of maintenance requirements for the radio astronomy telescope disclosed the need for a servicer that could maneuver and dock at several places on the structure for maintenance. This could be a very common maintenance requirement for future large-structure space systems. The consideration of an EOTS attached to a Tug and controlled remotely through the Tug systems appeared to fit this requirement. Such a concept for a servicer for use with

other satellites offers other advantages. If the EOTS manipulator is compatible with the reach and task functions, the EOTS could perform the Approach 1 maintenance tasks analyzed in this study. If a single point servicing were called for, an unfueled EOTS would be used and would remain attached to the Tug throughout the mission. Use of EOTS would save most of the development costs of a new servicer design.

Preliminary EOTS configurations were analyzed for compatibility with the maintenance of the satellites considered in this study. Figure IVD-11 shows the wrist reach envelope of the EOTS with an eight foot manipulator. In this configuration, the EOTS is offset towards the side of the Tug face. By rotating the satellite, relative to the Tug, about the central docking mechanism, the manipulator can reach modules 360° around the satellite. This type of configuration would accommodate most satellites.

A peculiar situation exists with the EOGP (see Figure IVD-12). Here the servicer must reach modules located both inside the equipment rings and modules installed outside. Since the EOGP is open at both ends, the docking mechanism would need to be offset. In fact, two docking mechanisms, as shown in Figure IVD-12), would be needed at each end to allow the 8-foot manipulator to reach all modules.

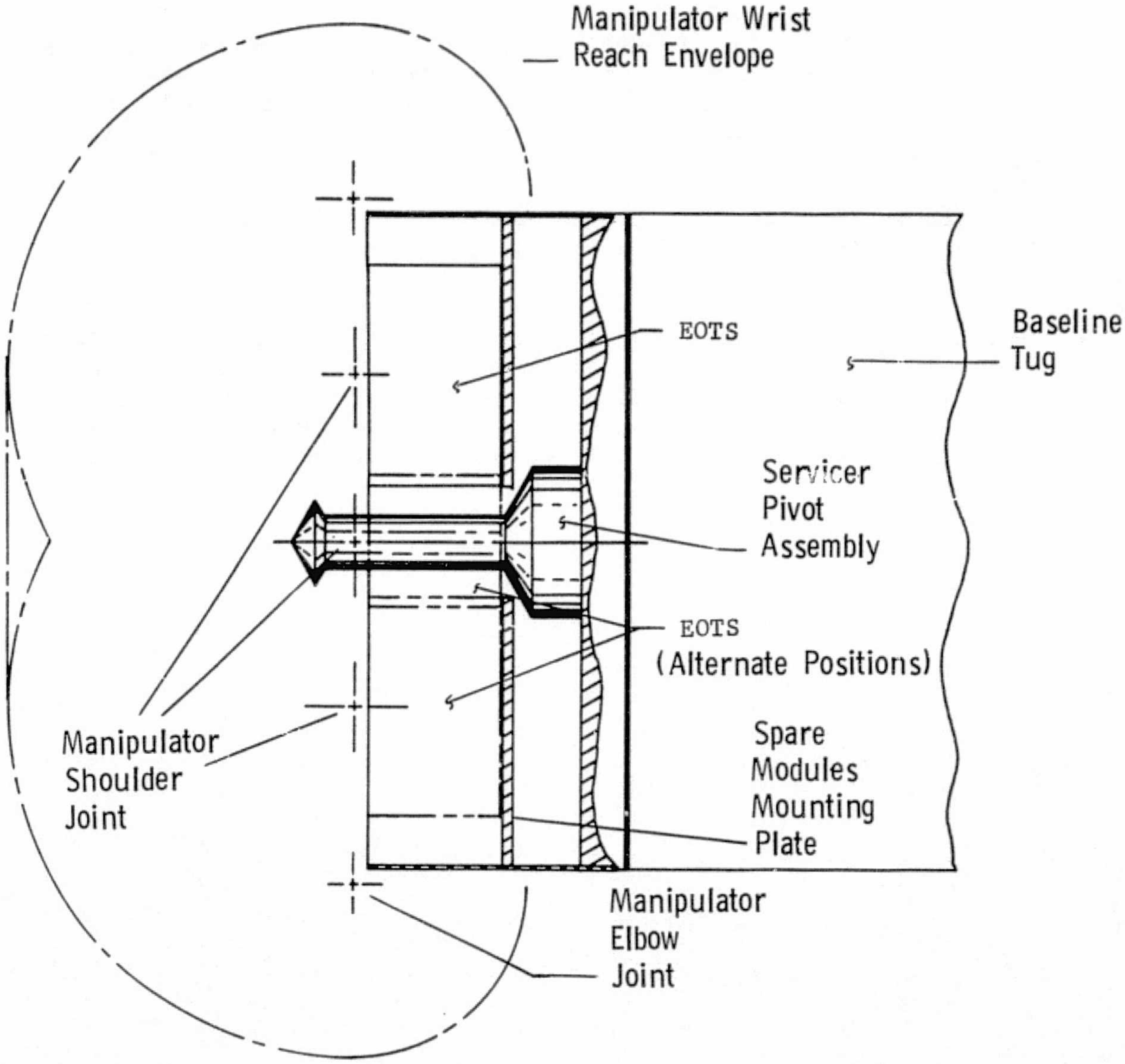

Figure IVD-11 EOTS Servicer Cross Section and Reach

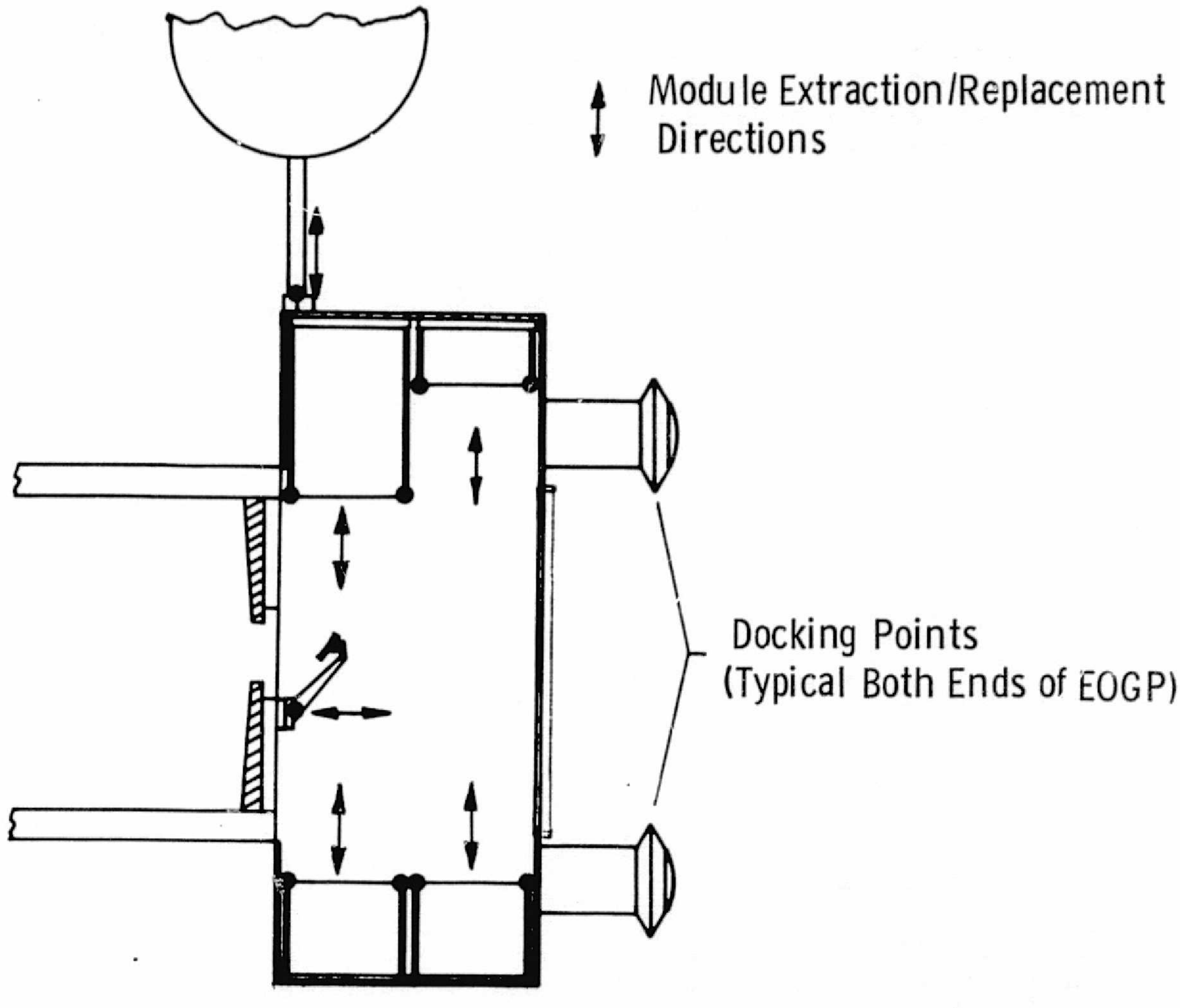

Figure IVD-12 EOGP Servicing

E. ON-ORBIT GEOSYNCHRONOUS MAINTENANCE VEHICLE (TASK 6)

The purpose of this task was to investigate the feasibility of an on-orbit automated maintenance vehicle that can remain in geosynchronous orbit for an extended period of time and carry equipment and spares to conduct maintenance, servicing, and refurbishment operations.

It is assumed that the vehicle is the RI version of the Solar Electric Propulsion Stage (SEPS), with an attached servicer. The servicer is assumed to contain docking provisions and a servicing system. The following weights are assumed:

SEPS		2967 lbs
Dry	2817.0	
Trapped fluids	15.0	
RCS propellant	135.0	
	2967.0	
Servicer[1]		950 lbs
Docking Mechanism	100.0	
Manipulator & TV Arm	400.0	
Structure	200.0	
Adapter	100.0	
Subsystems	150.0	
	950.0	
Hg Propellant-Max		2893 lbs

The SEPS, with full mercury propellant load, is capable of 625 days thrusting time. The on-orbit operational capability is 3 years, based on solar array degradation. Nominal thrust levels are 0.206 lb_f with an I_{sp} of 3000 seconds.

For longitudinal position changes (through elliptical-orbit phasing) with low-thrust systems, such as the SEPS, the following equations are used.[2]

---

1 Taken from RI Geosynchronous Platform Definition Study (SD73-SA-0036)

2 Based on equations from Propulsion Requirements for Controllable Satellites, ARS Journal, T. N. Edelbaum, August 1961, and information from E. Dazzo, Rockwell International, Seal Beach, California.

$$\Delta V = 4612.48 \sqrt{\frac{\phi}{W}}$$

$$\mu_p - 1 - e^{\left(\frac{-\Delta V}{96527.}\right)}$$

$$t = 0.1633\ \mu_p W$$

where:

ϕ	Phase angle change, degrees
W	Vehicle weight, lbs
ΔV	Velocity change, ft/sec
μ_p	Mass ratio, $\frac{W_f}{W_i}$
t	Time, days

(Assumes no shadowing or power degradation.)

Tables IVE-1 through IVE-4 presents the failure rate calculations for the replaceable modules on the four types of satellites. These data are used to determine the probable failure times and frequencies. Failure rates (λ)* (per module) are estimated based on information in the Aerospace Corporation Operations Analysis (Study 2.1), Payload Designs for Space Servicing, ATR-74 (7341)-3, June 1974 (and addendum, September 1974).

In addition to failure replacements (when whole numbers are exceeded in the failure rate tables), it is assumed that wearout items (solar arrays, power modules, ACS propulsion) are replaced at the AOT period.

If a SEPS servicing assembly were kept in geosynchronous orbit for a three-year period, it would need to contain module spares to enable exchange of the modules expected to fail or be depleted/degraded in that period. In addition, at least one spare is assumed available for any unique module. Table IVE-5 presents the spare allocations and servicing schedules, using the SEPS for three 3-year missions.

---

* λ given in hours to failure.

Table IVE-1 DWS Failure Rates

MODULE	TOTAL QTY EACH	UNIQUE QTY EACH	TOTAL QTY	TOTAL $\lambda \times 10^9$	FAILURES BY YEAR: 1	2	3	4	5	6	7	8	9	10
Solar Array Assy	2	1	2	36200	.4	.7	1	1.3	1.6					
TT&C	2	2	4	32000	.3	.6	.9	1.2	1.4	1.7	2.0	2.3	2.6	
ACS	1	1	2	26800	.3	.5	.8	1.0	1.2	1.5	1.7	1.9	2.2	
ACS Propulsion	2	1	2	22800	.3	.5	.7	.9	1.1					
Power Module	1	1	2	22400	.2	.4	.6	.8	1.0					
Amplifiers	2	2	4	24000	.3	.5	.7	.9	1.1	1.3	1.5	1.7	1.9	
Filters	2	2	4	24000	.3	.5	.7	.9	1.1	1.3	1.5	1.7	1.9	
Power Monitor	2	2	4	8000	.1	.2	.2	.3	.4	.4	.5	.6	.6	.8

Launch Schedule: 2 in 1982
AOT 5 years

Table IVE-2 Intelsat Failure Rates

MODULE	TOTAL QTY EACH SAT	UNIQUE QTY EACH SAT	TOTAL QTY	TOTAL λ x 10^9 EACH SAT	FAILURES BY YEAR: EQUIVALENT SATS * YEARS / YEARS								
					2/1	7/2	14/3	23/4	32/5	41/6	50/7	61/8	70/9
Solar Array Assy	2	1	1-9	12100	.3	.8	1.5	2.5	3.4	4.4	5.3	6.5	7.4
Transponders 35 of 48 reqd	8	4	8-36	12000	.3	.8	1.5	2.5	3.4	4.4	5.3	6.5	7.4
Power Module	4	4	4-36	33600	.6	2.1	4.1	6.8	9.4	12.0	14.7	17.9	20.3
TT&C 1 of 2 reqd	2	2	2-18	21500	.4	1.4	2.7	4.4	6.1	7.8	9.5	11.6	13.3
ACS 1 of 2 reqd	2	2	2-18	33400	.6	2.1	4.1	6.8	9.4	12.1	14.8	18.0	20.5
Receivers 1 of 4 reqd	2	2	2-18	12000	.3	.8	1.5	2.4	3.4	4.4	5.3	6.5	7.5
ACS Propulsion 2 of 4 reqd	4	2	2-18	22800	.5	1.4	2.9	4.7	6.5	8.3	10.0	12.1	13.8

Launch Schedule: 2 in 1983
3 in 1984
2 in 1985
2 in 1986
(6 operational)
AOT 6 years

Table IVE-3 SEOS Failure Rates

MODULE	TOTAL QTY EACH SAT	UNIQUE QTY EACH	TOTAL QTY	TOTAL $\lambda \times 10^9$ EACH SAT	FAILURES BY YEAR: EQUIVALENT SATS * YEARS / YEAR								
					1/1	2/2	3/3	4/4	6/5	8/6	10/7	12/8	14/9
Mission Equipment	2	1	1-2	7000	.1	.2	.2	.3	.4	.5	.7	.8	.9
Solar Array Assy	2	1	1-2	12100	.2	.3							
Power Module	2	2	2-4	16800	.2	.3							
ACS 1 of 2 reqd	2	2	2-4	25400	.3	.5	.7	.9	1.4	1.8	2.3	2.7	3.2
ACS Propulsion	2	2	2-4	22800	.2	.4							
Data Processing	1	1	1-2	17000	.2	.3	.5	.6	.9	1.2	1.5	1.8	2.1
TT&C	1	1	1-2	9100	.1	.2	.3	.4	.5	.7	.8	1.0	1.2

Launch Schedule: 1 in 1983
1 in 1987
AOT 2 years

Table IVE-4 TDRS Failure Rates

MODULE	TOTAL QTY EACH SAT	UNIQUE QTY EACH	TOTAL QTY	TOTAL $\lambda \times 10^9$ ALL SATS	FAILURES BY YEAR: 1	2	3	4	5	6	7	8	9
S-band Array Electronics	1	1	3	3900	.1	.1	.1	.2	.2	.3	.3	.3	.4
LDR TX & TT&C #1	1	1	3	30000	.3	.6	.8	1.1	1.4	1.6	1.9	2.1	2.4
Solar Array Assy & S-band Beam	1	1	3	54300	.5	1.0	1.5	2.0	2.4	2.9	3.4	3.9	4.3
HDR/MDR	2	2	6	36000	.4	.7	1.0	1.3	1.6	2.0	2.3	2.6	2.9
ACS #1	1	1	3	36600	.4	.7	1.0	1.3	1.6	2.0	2.3	2.6	2.9
ACS #2	1	1	3	33600	.3	.6	.9	1.2	1.5	1.8	2.1	2.4	2.7
ACS Propulsion	2	1	3	34200	.4	.6	1.0	1.3	1.6	1.8	2.1	2.4	2.7
LDR TX #2 & TT&C	1	1	3	41400	.4	.8	1.2	1.6	1.9	2.2	2.6	2.9	3.3
Power Module	2	2	6	67200	.6	1.2	1.8	2.4	3.0	3.6	4.2	4.8	5.3
TDRS/GS	1	1	3	27000	.3	.5	.8	1.0	1.2	1.5	1.7	1.9	2.2
Electronics Module	1	1	3	2100					.1	.2	.2	.2	.3
HDR/MDR Antennas	2	1	3	18000	.2	.4	.5	.7	.8	1.0	1.1	1.3	1.5
TDRS/GS Antenna	1	1	3	18000	.2	.4	.5	.7	.8	1.0	1.1	1.3	1.5

Launch Schedule: 3 in 1983 (2 operational)
AOT 5 years

Table IVE-5 SEPS 3-Year Missions, Spares Allocations and Servicing Trips

1983-1985			1986-1988			1989-1991		
DWS			**DWS**			**DWS**		
#Mod	Item	Weight#	#Mod	Item	Weight#	#Mod	Item	Weight#
2	SAA (2-85)	526	2	SAA (2-87)	526	2	SAA (1-91)	526
2	ACS propulsion	200	2	SAA (1-88)	526	2	ACS prop (1-91)	200
1	Power	179	2	ACS prop (1-86)	200	2	ACS prop (1-91)	200
1	TT&C (1-85)	27	2	ACS prop (1-87)	200	1	TT&C	27
1	ACS (1-85)	130	4	Power (2-87)	716	1	CFA	90
1	CFA (1-86)	90	1	TT&C (1-89)	27	1	Filter	10
1	Filter (1-86)	10	1	ACS (1-89)	133	9		1053
1	Power Monitor	20	14		2325			
10		1182						

1983-1985			1986-1988			1989-1991		
INTELSAT			**INTELSAT**			**INTELSAT**		
3	SAA (3-85)	405	6	SAA (3-87)	810	9	SAA (2-89)	1215
6	ACS prop (2-85)	810	11	ACS prop (4-88)	1485	19	ACS prop (3-91) (3-90)	2565
4	Power (3-85)	260	8	Power (6-87)	520	24	Power (8-90)	1560
3	Transponders (3-85)	225	6	Transponders (2-87)	450	6	Transponders (2-90)	450
3	TT&C (3-84)	120	5	TT&C (3-87)	200	6	TT&C (3-90)	240
4	ACS (3-85)	300	8	ACS (6-87)	600	8	ACS (6-90)	600
1	Receivers (1-85)	48	3	Receivers (3-87)	144	3	Receivers (3-90)	144
24		2168	47		4209	75		6774

1983-1985			1986-1988			1989-1991		
SEOS			**SEOS**			**SEOS**		
#Mod	Item	Weight#	#Mod	Item	Weight#	#Mod	Item	Weight#
2	SAA	224	6	SAA	672	4	SAA	448
2	ACS prop (1-84)	342	6	ACS Prop (1-86) (2-88)	1026	4	ACS Prop (2-90)	684
2	Power	252	6	Power	756	4	Power	504
2	Mission Equip	408				2	ACS (2-91)	412
1	ACS (1-86)	206	18		2454	1	Data Proc (1-91)	61
1	Data Proc (1-87)	61				15		2109
1	TT&C (1-90)	85						
11		1578						

1983-1985			1986-1988			1989-1991		
TDRS			**TDRS**			**TDRS**		
2	SAA (1-85) (1-88)	154	1	SAA (1-86)	77	2	SAA (1-90)	154
2	ACS Prop (2-85)	112	4	ACS Prop (1-87) (2-88)	224	2	ACS Prop (2-91)	112
2	Power (2-85)	113				2	Power (2-90)	113
1	S-band Electronics	51	4	Power (2-87)	226	1	LDR TX (1-90)	37
1	HDR/MDR (1-85)	44	1	HDR/MDR (1-88)	44	1	HDR/MDR (1-91)	44
2	ACS (2-85)	70	2	ACS (2-88)	70	2	ACS (2-91)	70
1	LDR TX (1-85)	31	1	LDR TX (1-88)	31	1	LDR TX (1-91)	31
1	Electronics Mod.	41	13		672	1	TDRS/GS (1-90)	46
1	LDR TX (1-86)	37				2	HDR/MDR Ant.	66
1	TDRS/GS (1-86)	46				1	TDRS/GS Ant.	14
2	HDR/MDR Ant. (1-88)	66				15		687
1	TDRS/GS Ant. (1-88)	14						
17		779						

NOTE: Number of trips and year serviced shown in parentheses.

Since there would be many modules involved in a 3-year period, it would not be practical or even workable to carry all spares along with the SEPS/servicer. An assembly, such as that shown in Figure IVE-1, appears feasible and is proposed. This assembly contains 1 or more spares tiers. Each tier would hold in the order of 24 spares modules. The SEPS/servicer would carry along only the single tier needed for maintenance of a satellite at some other longitude in orbit. The remaining spares tiers would be maintained at a "home base" longitude (100^{o}W assumed) by a stabilization unit.

The stabilization unit is assumed to weigh 400 lbs and each spares tier to weigh 200 lbs. Tables IVE-6, IVE-7 and IVE-8 present the analyses for using the SEPS/servicer to service the subject satellites during the three 3-year periods. The second and third missions require refueling the SEPS during the on-orbit period, using the RI proposed SEPS refueling unit.

The SEPS maintainance scheme would require 8 Shuttle/Tug flights if the SEPS are not recovered:

3 Tandem Tug flights to put SEPS/servicer/spares in orbit,
2 Tug flights for refueling SEPS.

Recovery of the SEPS/servicers would require three additional Shuttle/Tug flights. However, this would permit refurbishing the SEPS/servicer for later missions. Costs for both options will be investigated in tradeoffs, Section IV-F.

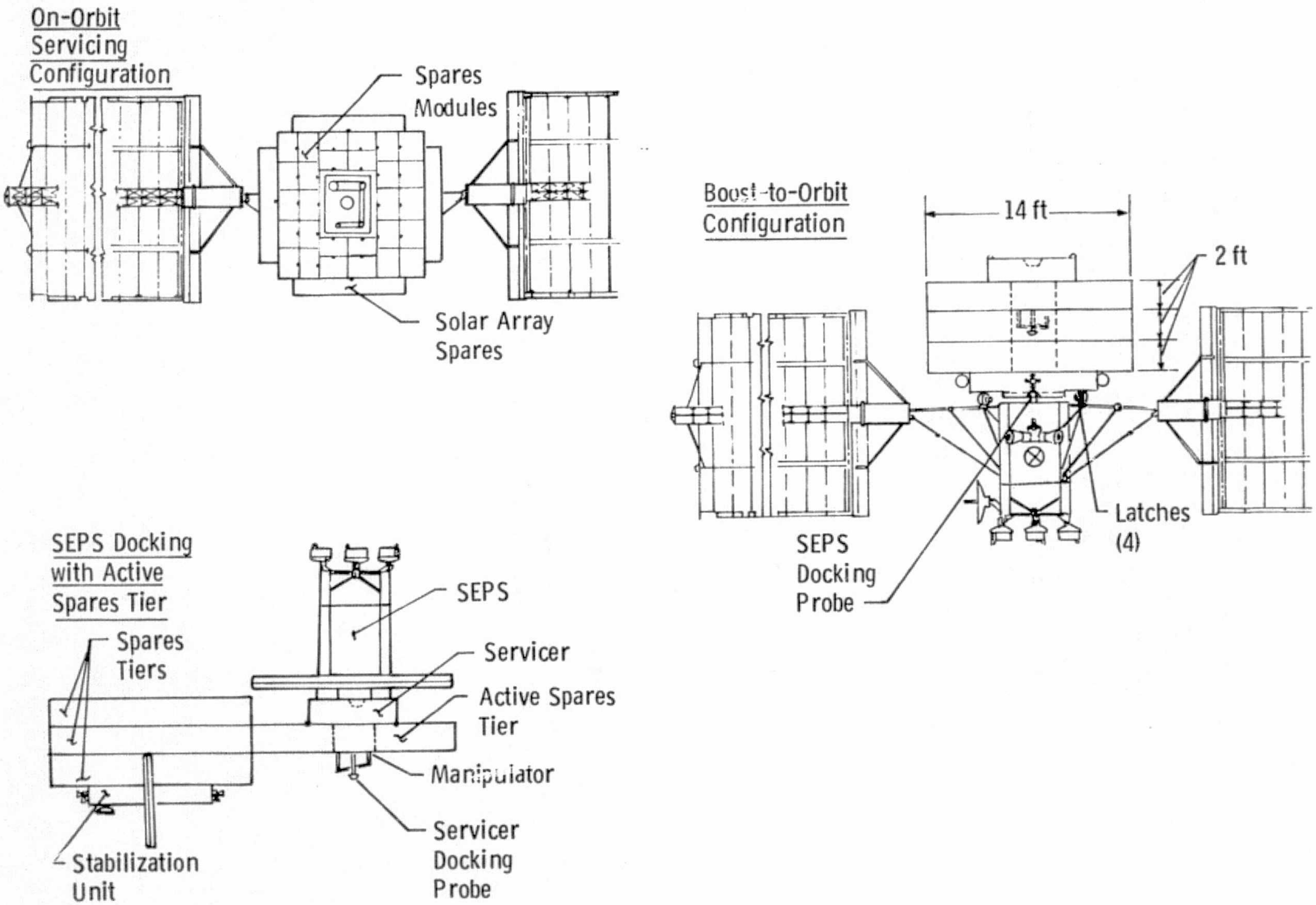

Figure IVE-1 SEPS/Servicer/Spares Assembly Configuration

Table IVE-6   1983-1985 SEPS Mission

SPARES TIERS	SATELLITES SERVICED	MODULES	SPARES WEIGHT	SERVICING TRIPS FROM 100°W HOME BASE
1	DWS/TDRS	27	1961	DWS 2 to 94°W, 2 to 124°W TDRS 5 to 41°W, 4 to 171°W
1	SEOS	11	1578	1 to 110°W
1	INTELSAT	24	2168	7 to 61°E, 5 to 174°E, 6 to 25°W

WEIGHTS -

SEPS	2967
Servicer	950
Pallet	200
Propellant	2893
	7010

TRIPS	ROUND TRIPS TO LONGITUDE:	$\Delta\phi$	SPARES	INITIAL TOTAL WEIGHT	ONE WAY ΔV	ONE WAY μ_p	ROUND TRIP W_p	ROUND TRIP t	FINAL WEIGHT
2-DWS	2 x 94°W	6	1961	8971	119.3	.001235	44.3	7.2	8926.7
1-SEOS	2 x 110°W	10	1578	8543.7	157.8	.00163	27.9	4.6	8515.8
7-INTELSAT	2 x 61°E	161	2168	9105.8	613.3	.0063	807.4	131.9	8298.4
5-TDRS	2 x 41°W	59	1961	8091.4	393.9	.0041	329.5	53.8	7761.9
2-DWS	2 x 124°W	24	1961	7761.9	256.5	.0027	82.4	13.5	7679.5
5-INTELSAT	2 x 174°E	86	2168	7886.5	481.7	.00498	392.5	64.1	7822.0
4-TDRS	2 x 171°W	71	1961	7287	455.3	.0047	274.3	44.8	7012.7
6-INTELSAT	2 x 25°W	75	2168	7219.7	470.1	.00486	420.9	68.7	6798.8
							2379.2	388.6 + 32 days servic- ing	

Servicing uses 2379# propellant

Servicing operations 421 days

Return SEPS weight 4431# (with servicer)

ΔV = 6527 ft/sec to 10,000 n.mi.

μ_p = .0654

W_p = 289.7#

t = 47 days

Propellant margin 224#

SEPS/Servicer/Spares/Stabilization Unit

Launched to geo using Tandem Tugs

SEPS	2967		
Servicer	950	1st Tug	56637
Stab Unit	400	2nd Tug	56458
Pallets (3)	600		113095
Propellant	2893		13517
Spares	5707		126612
	13517#		

Table IVE-7 1986-1988 SEPS Mission

SPARES TIERS	SATELLITES SERVICED	MODULES	SPARES WEIGHT	SERVICING TRIPS FROM 100°W HOME BASE
1	DWS/TDRS	27	2997	DWS 4 to 94°W, 3 to 124°W TDRS 6 to 41°W, 5 to 171°W
1	SEOS	18	2454	2 to 110°W, 1 to 100°W
1	INTELSAT	24	2109	4 to 61°E, 4 to 174°E, 5 to 25°W
1	INTELSAT	23	2100	5 to 61°E, 5 to 174°E, 4 to 25°W
1983 Pallet	DWS/TDRS	27	1961	DWS 1 to 94°W, 1 to 124°W TDRS 2 to 41°W, 2 to 171°W
1983 Pallet	SEOS	11	1578	1 to 110°W, 1 to 100°W

					ONE WAY		ROUND TRIP		
TRIPS	ROUND TRIPS TO LONGITUDE	$\Delta\phi$	SPARES	INITIAL TOTAL WEIGHT	ΔV	μ_p	W_p	t	ΣW_p
4-DWS	2 x 94°W	6	2997	10007.	112.9	.00117	93.6	15.3	93.6
6-TDRS	2 x 41°W	59	2997	9913.4	355.8	.00368	437.7	71.5	531.3
2-SEOS	2 x 110°W	10	2454	8932.7	154.3	.0016	57.1	9.3	588.4
4-INTELSAT	2 x 61°E	161	2109	8530.6	633.7	.0065	446.5	72.9	1034.9
1-DWS	2 x 94°W	6	1961	7936.1	126.8	.0013	20.8	3.4	1055.7
2-TDRS	2 x 41°W	59	1961	7915.3	398.2	.0041	130.3	21.3	1186.0
5-INTELSAT	2 x 61°E	161	2100	7924.	657.5	.0068	537.9	87.8	1723.9
1-SEOS	2 x 110°W	10	1578	6864.1	176.1	.0018	25.0	4.1	1748.9
3-DWS	2 x 124°W	24	2997	8258.1	248.7	.0026	127.5	20.8	1876.4
5-TDRS	2 x 171°W	71	2997	8130.6	431.0	.0044	362.2	59.1	2238.6
4-INTELSAT	2 x 174°E	86	2109	6880.4	515.7	.0053	293.3	47.9	2531.9
SEPS REFUELED									
5-INTELSAT	2 x 174°E	86	2100	9110	448.2	.0046	422	68.9	422.
1-DWS	2 x 124°W	24	1961	8549	244.4	.0025	43.2	7.1	465.2
2-TDRS	2 x 171°W	71	1961	8505.8	421.4	.00436	148.2	24.2	613.4
5-INTELSAT	2 x 25°W	75	2109	8505.6	433.1	.0045	380.8	62.2	994.2
4-INTELSAT	2 x 25°W	75	2100	8115.8	443.4	.0046	297.6	48.6	1291.8

Return SEPS weight 5803#

ΔV = 6527 ft/sec to 10,000 n.mi

μ_p = .0654

W_p = 379#

t = 62 days

Propellant Margin 1507#

Weight launched to geo using Tandem Tugs

SEPS	2967
Servicer	950
Stab Unit	400
Pallets (4)	800
Propellant	2893
Spares	9660
	17670#

NOTE: This hardware is split up in two Shuttle Orbiters. Assembled in Shuttle orbit when joining Tandem Tugs. Total P/L 130.765# is too much. SEPS propellant is off-loaded!!

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Table IVE-8 1989-1991 SEPS Mission

SPARES TIERS	SATELLITES SERVICED	MODULES	SPARES WEIGHT	SERVICING TRIPS FROM 100°W HOME BASE
1	DWS/TDRS	24	1740	DWS 2 to 94°W, 1 to 124°W TDRS 6 to 41°W, 5 to 171°W
1	SEOS	15	2109	2 to 110°W, 3 to 100°W
1	INTELSAT	25	2258	3 to 61°E, 3 to 174°E, 4 to 25°W
1	INTELSAT	25	2258	4 to 61°E, 3 to 174°E, 3 to 25°W
1	INTELSAT	25	2258	3 to 61°E, 4 to 174°E, 3 to 25°W
1985 Pallet	DWS	27	2997	1 to 94°W, 1 to 124°W
1983 Pallet	SEOS	11	1578	1 to 110°W

					ONE WAY		ROUND TRIP		
TRIPS	ROUND TRIPS TO LONGITUDE:	$\Delta\phi$	SPARES	INITIAL TOTAL WEIGHT	ΔV	μ_p	W_p	t	ΣW_p
2-DWS	2 x 94°W	6	1740	6822	136.8	.0014	38.6	6.3	38.6
6-TDRS	2 x 41°W	59	1740	6783.4	430.2	.0044	362.0	59.1	400.6
2-SEOS	2 x 110°W	10	2109	6790.4	177.0	.0018	49.8	8.1	450.4
3-INTELSAT	2 x 61°E	161	2258	6889.6	705.0	.0072	300.9	49.1	751.3
1-DWS	2 x 94°W	6	2997	7327.7	132.0	.0014	20.0	3.3	771.3
1-SEOS	2 x 110°W	10	1578	5888.7	190.1	.00197	23.2	3.8	794.5
1-DWS	2 x 124°W	24	1740	6027.5	291.1	.003	36.3	5.9	830.8
1-DWS	2 x 124°W	24	2997	7248.2	265.4	.0027	39.8	6.5	870.6
	SEPS REFUELED FULLY								
4-INTELSAT	2 x 61°E	161	2258	9268	607.9	.0063	465.5	76.0	465.5
3-INTELSAT	2 x 61°E	161	2258	8802.5	623.8	.0064	340.2	55.6	805.7
5-TDRS	2 x 171°W	71	1740	7944.3	436.0	.0045	358.1	58.5	1163.8
3-INTELSAT	2 x 174°E	86	2258	8104.2	475.1	.0049	238.8	39.0	1402.6
3-INTELSAT	2 x 174°E	86	2258	7865.4	482.3	.0050	235.2	38.4	1637.8
4-INTELSAT	2 x 174°E	86	2258	7630.2	489.7	.0051	308.9	50.4	1946.7
4-INTELSAT	2 x 25°W	75	2258	7321.3	466.8	.0048	282.6	46.1	2229.3
3-INTELSAT	2 x 25°W	75	2258	7038.7	476.1	.0049	207.8	33.9	2437.1
3-INTELSAT	2 x 25°W	75	2258	6830.9	483.3	.0050	204.7	33.4	2641.8

Return SEPS weight 4258#

ΔV = 6527 ft/sec to 10,000 n.mi
μ_p = .0654
W_p = 278#
t = 45 days
Propellant Margin 63#

Weight launched to geo using Tandem Tugs

SEPS	2967	1st Tug	56637
Servicer	950	2nd Tug	56458
Stab Unit	400		113095
Pallets (5)	1000		16905
Propellant	965		130000#
Spares	10623		
	16905		

F. TRADEOFFS (TASKS 3 AND 4)

1. Maintenance Costs

a. General - Cost estimates will be developed in this section for the maintenance programs using the on-orbit geosynchronous maintenance vehicle (Section IV.E) and the three maintenance approaches (Section IV.D).

It is assumed that there is no cost sharing of the Shuttle/Tug flights. These launch and mission costs are assumed to be \$11.2M per flight.

It is estimated that there will be maintenance missions to service 120 satellites (all serviceable satellites). Therefore, all maintenance-related non-recurring costs will be amortized over 120 servicings. There will be 20 servicings (through 1992) of the four satellites considered in this analysis.

The delta costs for developing serviceable configurations of the DWS, TDRS, Intelsat and SEOS are estimated by the following:

$$\triangle = \frac{0.1 \text{ X Expendable satellite RDTE costs}}{\text{Number of satellites}} \text{ X } \begin{matrix}\text{Estimated}\\\text{Complexity}\\\text{Factor (CF)}\end{matrix}$$

These delta costs are for such things as retractable appendages, docking aids, remote control circuits, and modular packaging. Costs of the expendable versions are estimated based on various sources and discussions. Table IVF-1 presents these delta costs for all the maintenance concepts.

Certain costs arise from special needs on the Shuttle and/or Tug to accommodate the maintenance approaches. Such needs include communications and data relay circuits, computer control of the servicer, man-rating requirements, and retractable airlock tunnel. Estimated delta costs for these requirements are listed in Table IVF-2. The costs are amortized over 120 maintenance missions for a per flight cost.

Table IVF-1 Delta Costs for Serviceable Satellites

SATELLITE (NUMBER)	RDTE COST $M	.1 * RDTE / NUMBER SATELLITES	MAINTENANCE APPROACH COSTS PER SATELLITE					
			SERVICER		MSM		RETRIEVAL	
			CF	COSTS,$M	CF	COSTS,$M	CF	COSTS,$M
DWS (3)	75	2.5	1.1	2.75	1.2	3.0	1.25	3.12
TDRS (4)	50	1.25	1.1	1.38	1.2	1.5	1.3	1.625
Intelsat (9)	90	1.0	1.1	1.1	1.2	1.2	1.2	1.2
SEOS (2)	130	6.5	1.0	6.5	1.1	7.15	1.1	7.15

Table IVF-2 Delta Costs to Shuttle/Tug

	Costs
On-orbit Maintenance Vehicle	$15.6M
Tug/Servicer	$20M
Manned Servicing Module	$40M
Tug Retrieval	$10M

Total program costs will be estimated for the four satellites so that the various maintenance-related costs can be viewed in perspective to the total costs.

Satellite procurement and recurring costs are estimated in Table IVF-3.

Table IVF-3 Satellite Procurement and Recurring Costs

	No. Procurred	Unit
DWS	3	$24M
TDRS	4	$18M
Intelsat	9	$23M
SEOS	2	$26M

It is assumed the satellites are launched per the following allocation:

2 DWS - 1 launch
3 TDRS - 1 launch
9 Intelsat - 5 launches
2 SEOS - 2 launches

b. On-orbit Geosynchronous Maintenance Vehicle - Cost data for the SEPS are taken from the RI SEPS studies (SD 74-SA-0176). The SEPS development costs are assumed split between the planetary and earth SEPS, i.e., $42.35M each. Recurring cost of the SEPS is $22.9M. Refurbishment costs are $4.0M. It is assumed, as RI proposed, that the SEPS test unit is refurbished for one of the flight units. Operational costs are $21.2M for a full mission.

Data on the SEPS refueling unit are based on information from W. Cooper, RI costs analyst on the SEPS studies. Non-recurring costs are $1.25M and first-unit costs are $0.4M. Two units are assumed for this program. Refurbishment costs are assumed to be $0.12M.

Servicer costs are based on the RI Geosynchronous Platform studies (SD 73-SA-0036-7). Non-recurring costs are $54.0M and recurring costs are $14.8M. Refurbishment is assumed to cost $2.0M.

Costs for the stabilization units and spares tiers are estimated as $20M for non-recurring, $2.0M for unit costs, and $0.5M operational costs.

Spares and replacement modules costs are based on a ratio of the module weight flown to orbit to the satellite weight, times the satellite recurring costs.

A summary of the costs for the four satellite programs using the SEPS 3-year maintenance scheme is presented in Table IVF-4.

c. Tug/Servicer Maintenance - Costs for the four satellite programs using the Tug/Servicer maintenance mode are presented in Table IVF-5.

The servicers are assumed refurbished after each mission at $2M each. Two servicer units are assumed procurred. Because of more flights, servicer costs are greater in this mode.

Costs of the replacement modules are based on the ratio of replaced weight to the satellite weight, times the satellite recurring costs. These costs are reduced in the option to return the modules. In this option, portions of the returned modules are assumed refurbished and reused. For five

Table IVF-4 Program Costs for SEPS Maintenance Missions

	COSTS, $M	
	OPTION 1A RETRIEVE SEPS	OPTION 1B NOT RETRIEVE SEPS
Shuttle/Tug Maintenance Support Flights	123.2	89.6
Support Equipment:		
SEPS	37.96	56.86
Refueling Units	0.58	0.58
Stabilization Units	5.8	5.8
Servicer	27.8	40.6
SEPS/Servicer Operations	21.2	20.7
Satellite Δ Costs	36.67	36.67
STS Δ Costs	1.43	1.04
Satellite RDTE	345.00	345.00
Satellite Procurement and Recurring	403.0	403.0
Initial Launch of Satellites	100.8	100.8
Spares and Replacement Modules	244.85	244.85
TOTALS	$1348.29 M	$1345.50 M

Table IVF-5 Program Costs for Tug/Servicer Maintenance Missions

	COSTS, $M	
	OPTION 2A RETURN MODULES	OPTION 2B EXPENDED MODULES
Shuttle/Tug flights	212.8	123.2
Servicer	72.9	56.9
Tug/Servicer Operations	2.85	1.65
Satellite Δ Costs	36.67	36.67
STS Δ Costs	3.33	3.33
Satellite RDT&E	345.0	345.0
Satellite Procurement and Recurring	403.0	403.0
Initial Launch of Satellites	100.8	100.8
Replacement Modules	154.69	176.98
Contingency Spares	68.00	68.0
TOTALS	$1400.04 M	$1315.53 M

of the Intelsats and four of the SEOSs, half of the returned modules are refurbished at 0.3 of the module procurement cost. The other returned modules are assumed not refurbished. Because of the late time in the program, the designs may change and/or the program might be about to expire (for those satellite models), thus not justifying module refurbishment.

It is assumed that a certain quantity of contingency spares will be available but not used in any mission. A cost of $68M was assumed to give a total replacement and spares module cost in the expendable module mode which is comparable to the cost of modules for the SEPS/servicer mode.

d. Manned Servicing Module Maintenance - Costs for the MSM maintenance mode are presented in Table IVF-6.

Table IVF-6 Program Costs for MSM Maintenance Missions

	COSTS, $M	
	OPTION 3A RETURNED AND REFURBISHED MODULES	OPTION 3B EXPENDABLE MODULES
Shuttle/Tug Flights	246.4	246.4
MSM	73.61	73.61
MSM Operations	1.5	1.5
Satellite Δ Costs	40.1	40.1
STS Δ Costs	6.67	6.67
Satellite RDT&E	345.0	345.0
Satellite Procurement and Recurring	403.0	403.0
Initial Launch of Satellites	100.8	100.8
Replacement Modules	154.69	176.98
Contingency Spares	68.0	68.0
TOTALS	$1439.77 M	$1462.06 M

The costs for the MSM are based on data from the RI Geosynchronous Platform studies. The other costs are generated per previous discussions.

e. Satellite Retrieval Maintenance Mode - Table IVF-7. presents a summary of costs for the four satellite programs for the modes of retrieving the satellite for maintenance at the orbiter or returning it to the ground for refurbishment.

Table IVF-7 Program Costs for Satellite Retrieval

	COSTS, $M	
	OPTION 4A MAINTENANCE AT ORBITER	OPTION 4B GROUND REFURBISHMENT
Shuttle/Tug Flights	473.0	506.0
Satellite Δ Costs	40.98	40.98
STS Δ Costs	1.67	1.67
Satellite RDT&E	345.0	345.0
Satellite Procurement and Recurring	403.0	403.0
Initial Launch of Satellites	100.8	100.8
Replacement Modules	154.69	154.69
Additional Ground Refurbishment		86.39
Contingency Spares	68.0	68.0
TOTALS	$1587.14 M	$1706.53 M

It is generally felt that when a satellite is returned to the ground for maintenance, more than just failed or depleted modules will be serviced. General refurbishment, including design update, will probably occur. Refurbishment costs are assumed to be the replacement module costs of options 2 and 3 plus an additional 30% of the remaining satellite recurring costs. Replaced modules are assumed to be refurbished for later use.

f. Cost Summary - Table IVF-8 summarizes the total program costs for the four maintenance modes (with options). Although there are several gross estimates in these cost analyses that prevent specific conclusions, some general conclusions can be made.

There would be little cost difference in the SEPS maintenance mode whether the SEPS vehicle is recovered or discarded in space. Perhaps the SEPS should be used to deorbit the stabilization units/spares for disposal at the end of the particular satellite program.

Table IVF-8 Total Program Costs with Maintenance

MAINTENANCE MODE	OPTIONS	COSTS, $B
1. SEPS - Three 3-Year Missions	A. Retrieve SEPS B. Not Retrieve SEPS	1.348 1.346
2. Tug/Servicer - at AOT	A. Return Modules B. Expended Modules	1.400 1.316
3. MSM - at AOT	A. Return Modules B. Expended Modules	1.440 1.462
4. Satellite Retrieval - at AOT	A. Orbiter Maintenance B. Ground Refurbishment	1.587 1.707

Considerable savings in STS flights and net costs could accrue from discarding replaced modules, with the Tug/servicer maintenance mode. However, this procedure would create much more space litter. This maintenance mode does appear to be the most economical method of maintenance.

The manned servicing module (MSM) method of maintenance is competitive with the other methods when more than one satellite can be serviced on one mission. Previous analyses which assumed single-satellite maintenance did not fully use the excess capacity of the Tandem Tugs and resulted in high program costs.

Retrieval of satellites from geosynchronous orbit for maintenance at the orbiter appears to be more costly relative to the other methods. Return of the satellites to the ground for refurbishment would be even more costly. However, this would permit more thorough updating of the satellite technology and capabilities.

2. Maintenance Mode Comparisons

a. Subjective Comparison - A comparative evaluation of the three maintenance approaches (excluding the on-orbit vehicle at this time) is presented in Table IVF-9. This table tabulates weighted ratings of ten factors considered in evaluating the maintenance approaches. The evaluation factors are given a weight-value designating their importance in the maintenance approach considerations. Each factor is given a unit rating for each approach, within a range of 1 to 10. The lowest value is considered the best. A value of 10 indicates a complete lack of capability for that factor. Intermediate values are assigned subjectively. The total rating of a factor is obtained by multiplying the weight-value times the unit-rating.

Man Safety - This factor considers the hazards to the Shuttle crewmen caused by the maintenance approach. Hazards involved in a nominal Shuttle mission are not considered. Experiences from past space programs have identified hazards and safety measures well. However, this must still be a strong evaluation factor.

Equipment Safety - This factor considers the possibilities of damage to the satellite equipment induced by the servicing operations and equipment.

Satellite Mechanical Complexities - This factor considers those elements on the subject satellite that must function to permit the completion of the maintenance mission, e.g. refoldable antennas and solar arrays for return to Shuttle orbit and/or the ground.

Servicing Approach Reliability - This factor considers those items that affect the success of the maintenance tasks such as aligning and mating module connectors and latches. The capability to overcome structural/functional deviations and general improvised repairs would be advantageous.

Table IVF-9 Subjective Evaluations of Maintenance Approaches

EVALUATION FACTORS	WEIGHT	APPROACHES					
		1 SERVICER		2 MSM		3 RETRIEVAL	
		UNIT RATING	TOTAL RATING	UNIT RATING	TOTAL RATING	UNIT RATING	TOTAL RATING
A. MAN SAFETY	15	1	15	5	65	4	60
B. EQUIPMENT SAFETY	14	5	70	2	28	1	14
C. SATELLITE MECHANICAL COMPLEXITIES	12	2	24	5	60	9	108
D. SERVICING APPROACH RELIABILITY	10	9	90	2	20	1	10
E. MISSION COMPLEXITY	8	5	40	6	48	5	40
F. SERVICING SYSTEM COMPLEXITIES	7	8	56	8	56	2	14
G. CAPABILITY OF TUG TO PERFORM MAINTENANCE MISSION	7	7	49	2	14	1	7
H. CAPABILITY OF RETURNING SATELLITE TO GROUND	6	10	60	10	60	1	6
I. AVAILABILITY OF ORBITER FOR OTHER ACTIVITIES	4	1	4	3	12	6	24
J. DEVELOPMENT PROGRAMS	3	6	18	8	24	2	6
TOTALS			426		387		289

NOTE: LOWEST VALUES ARE THE BEST.

Mission Complexity - This item considers the overall mission activites such as the quantity and complexities of the Shuttle and Tug flights, ground support operations, and communications and remote-control requirements.

Servicing System Complexities - This item considers support equipment complexities and reliability from the standpoint of the possible failure to accomplish required maintenance activities. For example, preprogrammed control of the servicer is desirable but a backup remote-control capability would maintain a high maintenance reliability at the expense of time. Approach 2 EVA activities would be reliable but the complexity of the MSM systems must be considered.

Capability of Tug to Perform and Complete Required Maintenance Mission - This item considers the Tug and support equipment performance capabilities. Limitations on Tug capabilities would limit the stay-time in orbit. Contingencies that extend or delay maintenance operations might result in an abort before completion of the maintenance tasks.

Capability of Returning the Satellite to the Ground for Refurbishment - This factor considers the advantage inherent in approach 3 where a satellite is at the orbiter for maintenance and therefore could be returned to earth for depot level maintenance without additional Shuttle transportation costs. To redeploy the satellite in its operational orbit requires an additional STS launch(s) in either case.

Availability of Orbiter for Other Activities - This factor considers the probability of the orbiter being used in other experiment activities while the maintenance activities are in process. Certainly, while the maintenance equipment is away from the orbiter, the orbiter can be used in other activities. However, the particular orbit may not be compatible with other experiments. This would be more disadvantageous when several orbiters are used in the maintenance mission and are restrained to a particular locale. This item is more of an economic factor because of possible payload cost sharing but is evaluated subjectively because specific missions cannot be defined at this time.

Development Programs - This factor considers the magnitude, excluding the economics, of developing and integrating the servicer, MSM, or any other support equipment peculiar to the approach.

b. On-Orbit Geosynchronous Vehicle Comparison - The operation of the on-orbit maintenance vehicle would be similar to the Tug/servicer maintenance mode. However, the on-orbit vehicle would rate worse because of the need for long duration reliability, the multitude of orbit phase changes, and the overall complexity of supporting equipment and mission operations.

3. Conclusions and Recommendations

In general, costs of the various maintenance options are inversely proportional to the results of the subjective evaluations. Manned maintenance operations appear to be more desirable in spite of some safety hazards. However, costs of manned operations tend to be greater. The benefits of man in any maintenance operation cannot be forecast in any analytical evaluation. For instance, the need for manned repair capabilities on Skylab could not be predicted but the value of direct manned repair activities in those missions are now known to all. In the maintenance of satellites, many component failures can be predicted and mechanical means devised to effect most repairs. However, manned participation in maintenance activities become invaluable in those type of repairs where unpredicted failures occur which call for on-the-spot troubleshooting, inspections, and repairs of non-module type hardware. In the case of the satellites investigated in these studies, the following potential maintenance activities would be more feasible or appropriate for manned activities:

Repairs

- Broken wires
- Defective module attachment mechanisms
- Bent/defective pin connections
- Ripped/punctured antennas
- Fluid system leaks
- Frozen (contact weld) joints
- Replace fixed sensors
- Replace appendages not designed for changeout
- Attach thermal control coverings

Inspections

Electrical shorts
Bent or loose members
On-the-spot electrical circuit checks
Corrosion/wear points

The primary difference in the costs between the various maintenance options is the costs of the Shuttle/Tug flights. Boost vehicles and orbit-to-orbit vehicles of greater capacity could make the manned maintenance modes more attractive. This potential should be investigated in other studies.

V. FUTURE STUDY AREAS

Further studies are recommended to provide technical depth in key elements and to assess potentially important areas not analyzed in this study because of time and scope limitations.

1. Total Power Satellite Design - Perform an in-depth analysis of the structural and assembly techniques for the total Satellite Solar Power Station. This effort would be a continuation and expansion of the Raytheon/Grumman analysis and design, combined with the Martin Marietta MPTS design criteria. The study should include developing a total support structure for the solar cell array and the MPTS, with techniques for assembly. These analyses must be associated with a definition of the logistics techniques necessary to boost the components to HEO. Maintenance and contingency modes should also be considered.

2. Packaging Density Analysis - Investigage ways to increase the packaging density, of the structural components for large space structures. The MMC concept of collapsable beams, in pallets, fills the total volume of the shuttle cargo bay, however, only 62% of the weight capacity of the shuttle is used.

3. Space Logistics Analysis - Analyze logistics techniques for large space structures. Perform trade-offs on cost of boosters, time to boost to orbit, weights to be boosted and the altitude to which the structure is boosted. Derive ground rules for the most effective space transportation system utilization as related to particular classes of satellite. Determine the impact of the heavy lift shuttle orbiter on the concepts for structural components and their boost to orbit techniques.

4. Structural Commonality - Analyze logistics, assembly, and structural requirements for all proposed large space structures to derive a common or universal base structure and assembly approach. Evolve a building block approach which can be extended to meet the requirements of any of the large structures.

5. Manned Orbital Assembly - Investigate the use of man as a direct aid to HEO space system assembly and maintenance. Examine the cost vs reliability of man's presence. Identify man's long term needs in HEO, space station, crew cycling, transportation costs, man rating costs and complexities of systems he would interact with, etc.

6. Further MPTS Analysis - MMC has completed adequate analysis to show the feasibility of our proposed approach to the MPTS structural assembly. Further analysis and design are needed in the following areas:

a. Mobile assembler

b. Structural dynamics

c. Thermal control

d. Remote welding and bonding

e. Pyrotechnics for assembly

f. Video systems and lighting for assembly

g. Alignment devices

h. Maintenance of microwave transmission hardware

7. On-Orbit Fabrication Plant - To support on-going orbital assembly programs, it may be advantageous to establish a manned manufacturing facility in orbit. Transporting raw-stock building materials to the facility, with subsequent manufacture of the space system structural elements, would utilize the STS more efficiently. Expended orbiter external tanks offer a potential source of raw materials that could be used in the manufacturing facility. These applications should be analyzed for potential benefits to orbital assembly programs.

8. Low Earth Orbit Demonstration of Assembly Techniques - A natural progression from the results of this study is the demonstration of assembly techniques in orbit. The first steps should be the development and demonstration of assembly tasks critical to any large space structure assembly. Such tasks include fastening techniques, alignment methods and hardware, orbital dynamics and stabilization requirements, etc. Following this, an actual assembly of a useful space system would be a logical follow-on.

9. SEPS Maintenance Program Reliability - The use of an on-orbit vehicle for continuous availability to perform replacement maintenance on geosynchronous satellites appears to offer economic benefits. However, the reliability of the propulsive

vehicle and associated support equipment should be analyzed further to determine the reliability of the systems over the duration of the planned maintenance period. This study considered the use of a SEPS, a new spares pallet stabilization unit, and SEPS refueling units in three-year maintenance periods.

10. Low-Thrust Boost Vehicles - The use of low-thrust propulsive vehicles offers advantages for boosting large space structures to higher orbits. However, vehicles dependent on solar energy, such as the SEPS, experience solar cell degradation in low-orbit radiation environments and loss of propulsive power when in earth shadow. Low-thrust vehicles not hindered by these problems (such as nuclear propulsion types), should be investigated for application to inter-orbit transportation.

APPENDIX A

SUPPORT EQUIPMENT

As a part of Task 1, preparatory data were compiled on various items of support equipment that might be used in the assembly and maintenance tasks. Support equipment that is presently baselined in the Space Shuttle Program or is being considered is discussed.

1. Shuttle Orbiter

The orbiter (Figure A-1) provides a 15-foot diameter by 60-foot long payload bay and can deliver and rendezvous a 65,000 lb payload to (and retrieve 32,000 lbs from) a 160 nautical mile circular orbit at 28.5 deg inclination (assuming a KSC launch). Internal orbital maneuvering subsystem (OMS) tanks provide the capability for 1000 ft/sec ΔV. The nominal on-orbit duration is seven days for a four-man crew, but can be extended to 30 days with added mission extension kits, chargeable to the payload.

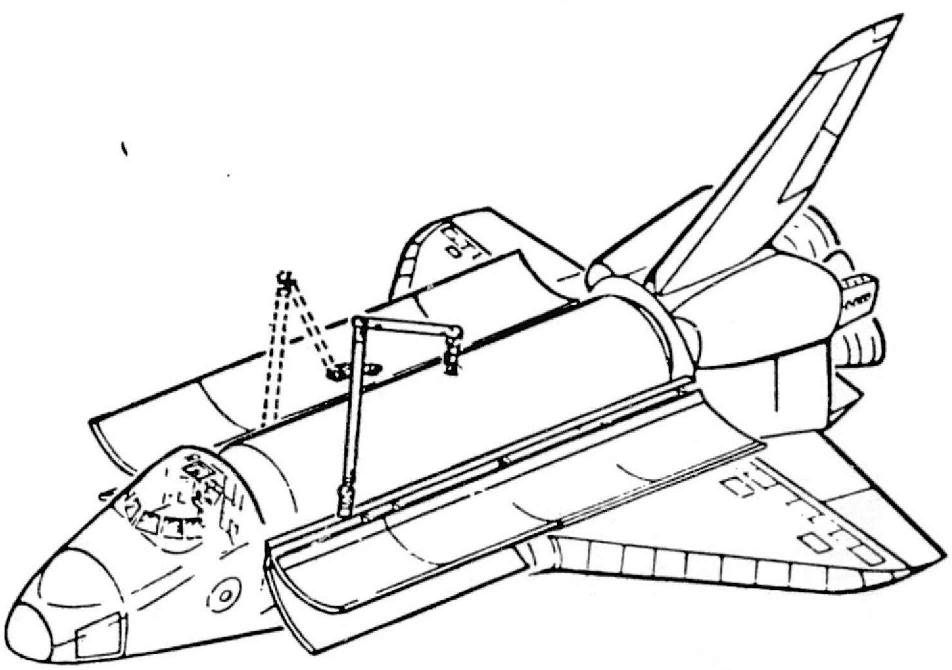

Figure A-1 Shuttle Orbiter

Maximum design accelerations are:

Landing	1.50 g
Crash	46 g (11 millisecond duration)
Boost	3.3 g (X-axis)

Design vibration limits are:

$0.06\ g^2/Hz$ at 75 to 300 Hz (± 3 dB/octave roll-off)

2. Interim Upper Stage (IUS)

The IUS (Figure A-2) provides a third stage capability to the Shuttle Transportation System (STS) for boosting satellites from the Shuttle orbit to higher orbits. The IUS basically uses existing U. S. booster vehicle hardware, but with additional development work needed. The IUS is planned for use in the 1980 to 1984 time frame. Two IUS operational modes are considered:

a. Expendable version for delivery-only operations with no rendezvous or TV and data storage capabilities;

b. Reusable version using a Transtage with a kick stage.

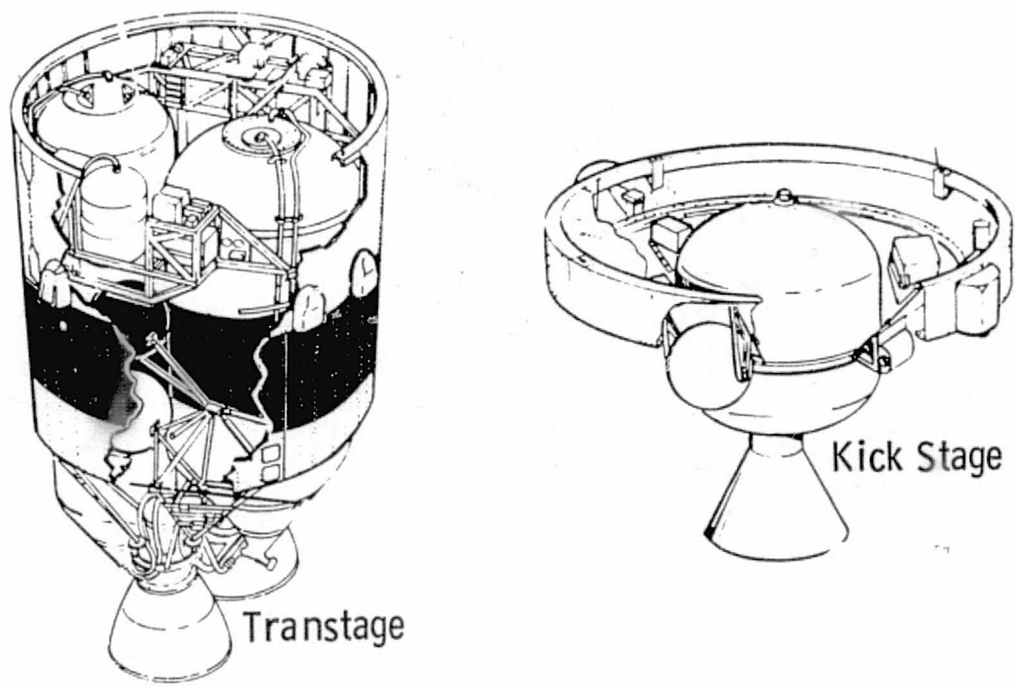

Figure A-2 Interim Upper Stage

Comparative data of the two versions are:

	Expendable	Reusable
Thrust, lbs	16,000	16,000 and 15,000 for kick stage
I_{sp}, sec	311	311, Transtage; 288, kick stage.
Payload to synchronous orbit, lbs	5,700	3,900
Size, ft	10 diameter x 19.1 long	10 diameter x 25 long
Weight (loaded), lbs	36,823	42,128
Weight (dry), lbs	4,407	5,312

3. Tug

The Tug (Figure A-3) provides a third stage capability to the STS for boosting satellites from the Shuttle orbit to higher orbits and/or retrieving satellites. The Tug is considered for use in the post-1984 time. Because the Tug was used more extensively in this study, more emphasis was placed on understanding the Tug capabilities. The following data and information were extracted from Baseline Space Tug Configuration Definition, MSFC 68M00039-2 and Baseline Space Tug Flight Operations, MSFC 68M00039-3, July 1974.

For geosynchronous missions the Tug will be used to 1) transfer a spacecraft (SC) from the Shuttle and deploy it in geosynchronous orbit, 2) deploy and retrieve (also multiple deploy) SC, or 3) retrieve SC from geosynchronous orbit.

The Tug transfers from the Shuttle orbiter (160 x 160 n.mi circular earth orbit at 28.5^{o} inclination) to geosynchronous orbit through the following steps (see Figure A-4):

1) As the Tug approaches a mode (ascending or descending), it ignites and optimally transfers onto a phasing orbit. The phasing orbit is designed to allow the Tug to arrive at a specified target longitude in the synchronous orbit. Inclination is reduced the optimal amount during each burn.

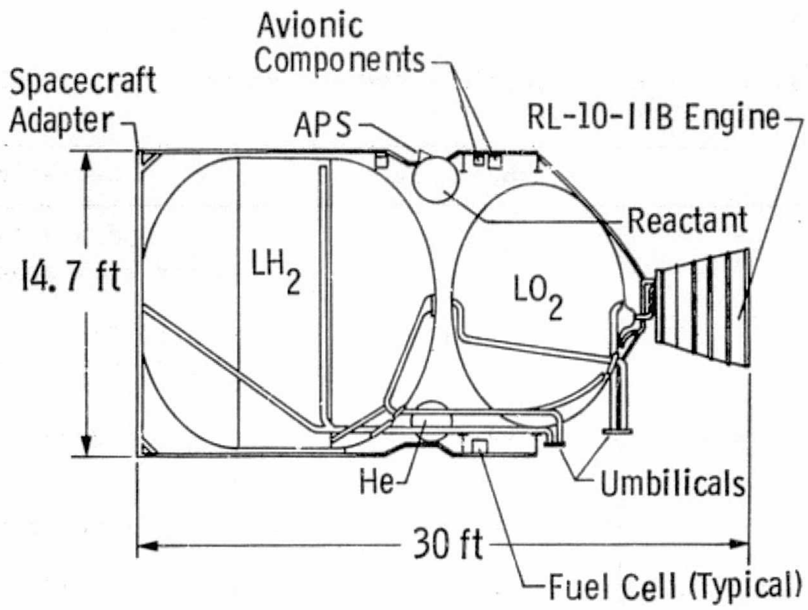

Figure A-3 Tug

2) After coasting for approximately one revolution in the phasing orbit, the Tug performs the second maneuver to place it on a transfer conic with an apogee near geo-synchronous altitude.

3) After coasting to apogee, the vehicle circularizes in the final orbit and deploys a SC or initiates rendezvous and docking with a SC.

Similar maneuvers are performed for return to the orbiter. The Tug returns to a 170 x 170 n.mi. orbit above the orbiter where rendezvous and docking is initiated. (See Figure A-5.) Final capture and docking is achieved by use of the orbiter RMS.

The Tub weight breakdown is summarized in Table A-1.

Main Propulsion System (MPS) - The characteristics of the MPS are:

	Full Thrust	Maneuver Thrust (Pump Idle)	Tank Head Idle
Steady State Thrust, lb.	15,000	3,750	157
Specific Impulse, sec.	456.5	434.7	377

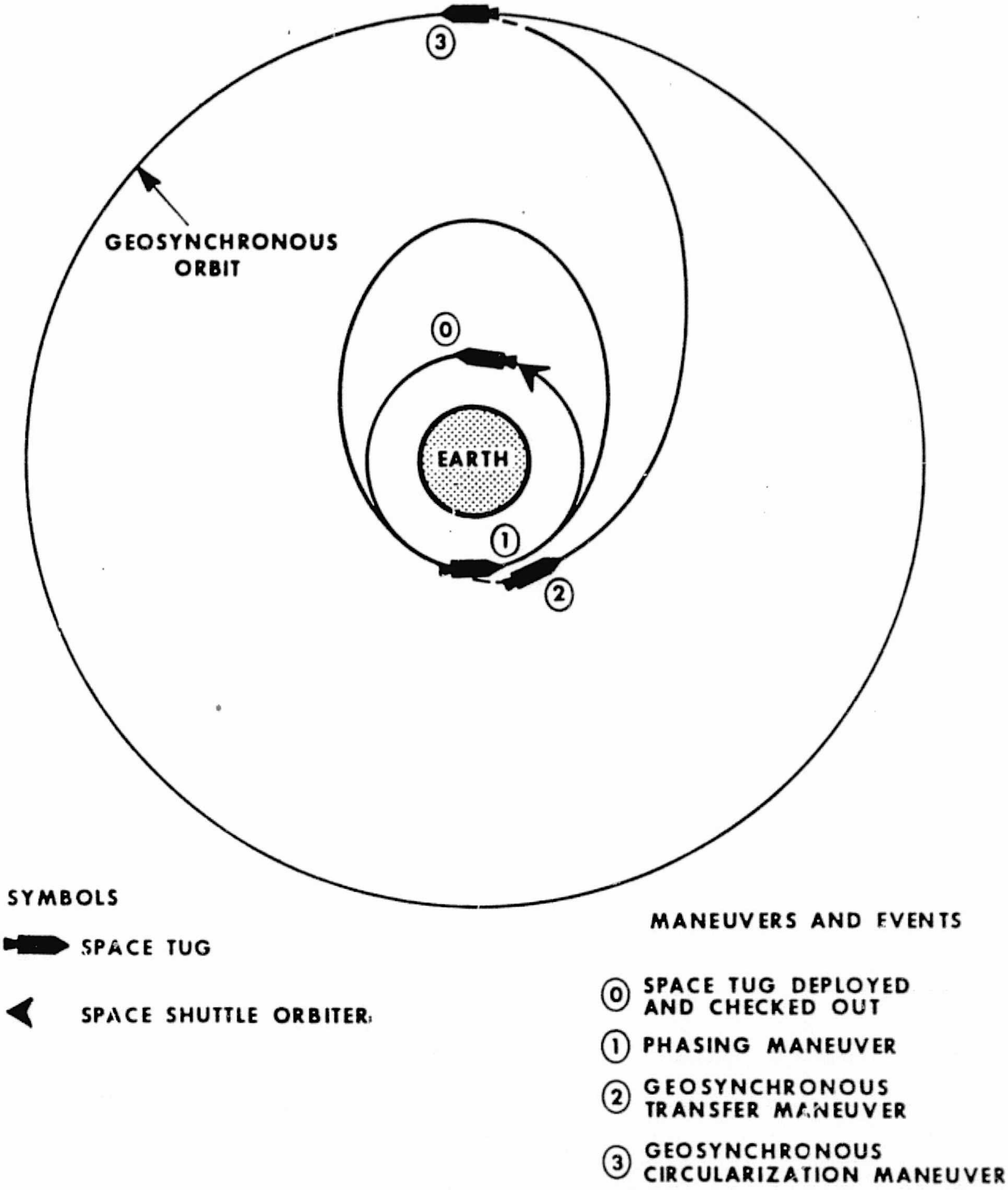

Figure A-4 Space Tug Geosynchronous Ascent Profile

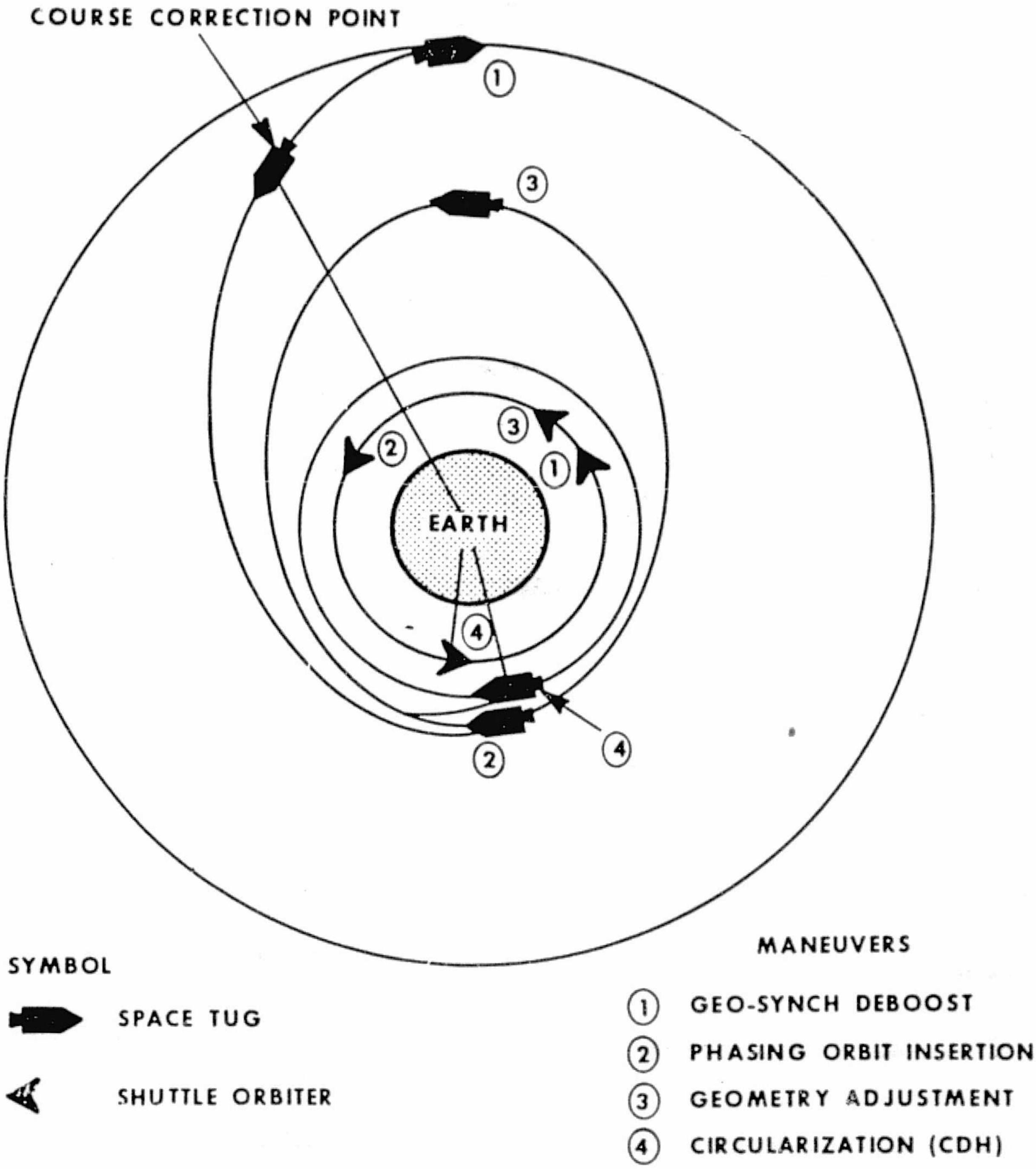

Figure A-5 Space Tug Geosynchronous Deboost Profile

DESCRIPTION	WEIGHT (LB)	
STRUCTURE		1974
Body Shell	914	
Fuel Tank & Supports	425	
Oxidizer Tank & Supports	243	
Thrust Structure	29	
Mounting Structure	100	
Payload & Umbilical Interface	263	
PROPULSION		1346
Engine	442	
Feed, Fill, Drain & Vent	256	
Pneumatic & Press	234	
Hydraulic	63	
Propellant Loading & Measuring	50	
APS	301	
THERMAL CONTROL		441
Active Thermal Control	70	
Fuel Tank Insulation	90	
Oxidizer Tank Insulation	40	
Insulation Purge	200	
Passive Thermal Control	41	
AVIONICS		921
Navigation Guidance and Control	154	
Data Management	158	
Communications	72	
Measuring System	92	
Electrical Power and Distribution	410	
Rendezvous & Docking	35	
10% GROWTH CONTINGENCY INCLUDING FASTENERS		468
TOTAL DRY WEIGHT		5150

DESCRIPTION	WEIGHT (LB)	
UNUSABLE RESIDUALS		576
Trapped Propellant	150	
Trapped Gases	330	
Fuel Bias	65	
Hydraulic Fluid	5	
APS Trapped	19	
Trapped Water	7	
BURN OUT WEIGHT (w/o APS Reserve)		5726
BURN OUT WEIGHT w/APS Reserve (varies with mission) Deploy only	29	5755
(Dual deploy + 1 Retrieve, 45)		
EXPENDABLES		547
LOX Boiloff	130	
Fuel Boiloff	165	
Start/Stop	77	
Fuel Cell Reactants	175	
PROPELLANT RESERVES		300
USABLE PROPELLANTS		50,177
LH_2	7127	
LOX	42,762	
APS	288	
FIRST IGNITION WEIGHT		56,779
ORBITER INTERFACE ACCOMMODATIONS AND BOTTLES (includes contingency)		1900
Adapter Structure	676	
Propulsion	178	
Dump Press	126	
Avionics	470	
JSC Fittings	450	
GROUND LIFT-OFF		58,679

Table A-1 Tug Detailed Weight Breakdown

The tank head idle (THI) mode is used to provide for engine/feed line childown and thrust to settle propellants in the main tanks. The pump idle mode is used as needed for small ΔV maneuvers.

Auxiliary Propulsion System (APS) - The APS provides the required impulse to position the Tug during coast periods and perform translational maneuvers as required for rendezvous and docking. The systems employ four clusters of six thrusters each mounted at 90^{o} intervals around the Tug. Each thruster gives 25 lbs. thrust. The steady state specific impulse is 230 sec. and the pulsing specific impulse is 160 sec.

Tug/SC Docking - The current baselined SC docking system is fully automated and consists of:

1) Docking mechanisms on the Tug and SC
2) Auxiliary propulsion system
3) Laser radar
4) Corner reflectors on the SC
5) Control software
6) Override television monitor function

The current accuracy requirements for the baseline design of the Tug and the SC relative rates and alignments are:

radial misalignment	1.0 ft.
misalignment angle	5.0 degrees
longitudinal closure	0.1 - 1.0 ft/sec.
lateral closure	0.3 ft/sec.
angular closure	0.5 deg/sec.

Most of the docking methods and requirements are under study.

Onboard Guidance and Navigation (G&N) Software - The guidance software will have the capability to maintain a stationkeeping position with respect to a target SC and issue those commands necessary to achieve docking with the target SC. The navigation software will provide laser radar scanning data to the guidance software during the stationkeeping and docking maneuvers for pattern recognition and rotation rate determination of the target SC. The onboard navigation software will have the capability to process TDRS data needed to achieve an onboard navigation update and accept a navigation update from the STDN for those orbits that are outside of the TDRS capability.

Onboard Flight Sequencer - The onboard flight sequences shall be the main onboard flight computer control logic in which all onboard events are initiated by the sequencing logic and all external (Orbiter or ground) commands to the Tug shall be processed through this sequencing logic. The onboard flight sequencer shall perform the following functions:

1) Contains the nominal onboard timeline of events and sets up the required time bases for achieving the mission.

2) Passes the control to the appropriate subsystem to execute the events of the timeline.

3) Receives the ignition times and propulsion system modes (APS, tank head idle, pump idle, and mainstage thrust) selected to execute the desired maneuver from the guidance software and generates the necessary discrete signals to activate the propulsion.

4) Generates the discrete signals necessary to activate the navigation alignment sequence in that particular navigation subsystem.

5) Receives and processes the data from the TDRS system and passes the data on to the navigation system.

6) Receives navigation updates from the STDN and routes it to the navigation system.

7) Accepts time corrections to the nominal sequence of events from the guidance software system.

8) Accepts signal from the ground to inhibit a planned power flight maneuver.

9) Accepts signals from the ground for target updates and alternate missions.

10) Accepts signal from the ground or guidance software system to start the laser radar.

11) Accepts signals from the ground to start TV system and execute scanning patterns.

12) Accepts signals from the Orbiter or ground to generate those discrete signals necessary to activate the safing system.

13) Accepts signals from the Orbiter or ground to execute attitude pointing.

14) Accepts caution and warning signals and generates the necessary signal discretes to take the appropriate Tug systems actions.

Mission Operations - To minimize overall operational costs of the STS, the Shuttle orbiter will be used to conduct other activities in low-earth orbit (LEO) while the Tug is away. Separate operations teams are employed. The Tug operations team will have responsibility for the Tug when separated from the orbiter. The primary Tug functions will nominally be onboard automated, with ground command for malfunction diagnosis, contingency operation, and preplanned functions for which real time analysis is required.

The Orbiter crew will have the prime responsibility for monitoring Tug systems which are crew-safety related. The Orbiter crew will have the responsibility for the deployment of the Tug/SC via the Orbiter deployment system and crew operated manipulators.

After Tug/SC release from the Orbiter and the Orbiter retirement to a safe distance, the Tug/SC ground operations team will verify systems readiness for intial Tug burn and will initiate configuration of the stage and propulsion systems for mission accomplishment. After Tug mission completion, the Tug/SC operations team will be responsible for placing the Tug/SC in the rendezvous orbit and performing Tug/SC systems and propellant tank safing in preparation for Orbiter recovery.

The Orbiter crew will then have the responsibility for the recovery and stowage of the Tug/SC. This will include the performance of the required Orbiter rendezvous terminal phase maneuvers (closing, braking and stationkeeping); the visual acquisition and physical capture of the Tug/SC; the fitting of the Tug/Orbiter interface connections (mechanical, electrical, and fluid); stowing the Tug/SC in the Payload bay; and the final evaluation/preparation of Tug/SC systems for entry.

During the time the Tug/SC is attached to the Orbiter the Tug/Shuttle operations teams will coordinate all mission activity which will have a direct influence on crew activity or Orbiter operations. Commands to the Tug and the SC will originate in the appropriate control facility and will be routed through the Shuttle Operations Center (SOC) for uplink to the Orbiter command system and subsequently to the Tug or SC. Tug and SC data will be downlinked through the Orbiter Telemetry System and will be separated at the STDN/TDRSS ground station for transmission to the appropriate operations facility.

During the period when the Tug/SC is not attached to the Orbiter the interface requirements change. The Orbiter is no longer a through put device for the Tug/SC command and telemetry. The crew has no responsibility for Tug operations, and the coordination with the Orbiter control center is essentially that of an exchange of information, e.g., ephemeris data. The Tug and SC systems data from the Tug are routed directly to the respective control centers from the TDRSS and STDN ground stations. Nominal and contingency commands are generated and initiated from the respective control centers; however, any commanding that may affect the operation or status of both vehicles must be coordinated prior to execution. There is no requirement at this time for coordination between the Shuttle and SC centers for this or subsequent mission phases. From Tug deployment through SC deployment and retrieval, the Tug is essentially independent of the Orbiter and utilizes a direct communications from the STDN/TDRSS to the Tug Operations Center (TOC). The TOC will continue to be independent of the SOC with the exception of the coordination interface (may require some data exchange) until Tug retrieval where again the Orbiter will provide all communications.

After the Tug/SC has been released and is in the vicinity of the Orbiter (i.e., could potentially pose a hazard to the Orbiter) the Orbiter center will provide the status of the Orbiter in relation to the Tug. The Tug center will inform the Orbiter center of the status of Tug on-orbit systems configuration and will maintain the Tug in a safe mode until the Orbiter has retired to a safe distance.

In order to effect Tug/SC and Orbiter rendezvous, intercenter mission planning will be necessary. The required targeting and orbital parameters must be developed and concurred in by both centers. Prior to Orbiter capture of the Tug vehicle, the Tug center will accomplish vehicle safing and will notify the Orbiter center that terminal rendezvous and retrieval may begin. The Tug center will monitor Tug retrieval and will assist, should unforeseen difficulties arise during this operation.

<u>Data Network</u> - The network for Tug operations will consist of the Spaceflight Tracking and Data Network (STDN) complemented by the Tracking and Data Relay Satellite System (TDRSS) with a ground terminal station at White Sands, New Mexico. Data transmission to and from the STDN and TDRSS is through the NASA Communications Network (NASCOM), a global network providing operational ground communications support.

The STDN is a worldwide complex of stations used to provide communications with both manned and unmanned spacecraft. Present plans are for STDN to consist of no more than eight stations with three basically for deep space support, two for launch support, and two or three more for special applications. The TDRSS will have two relay satellites at geosynchronous orbit 130 degrees apart with a third satellite spare on operational standby as a backup.

Real time operational control and scheduling of the networks are provided by the Network Operations Control Center (NOCC) located at the Goddard Space Flight Center (GSFC), Greenbelt, Maryland.

The STDN and/or TDRSS will be the prime network during the Tug operations that will provide tracking, telemetry and command support. This service will provide coverage of orbital operations for orbits below 5000 KM, for which the TDRSS will be the prime support system. For higher altitude orbits the deep space STDN stations will provide the primary support, with STDN continuous coverage from approximately 6500 KM. Both subnets will complement each other when required in providing full support.

The NASA Tug missions will be supported by the NASA networks. The DOD Tug missions will be supported by the DOD networks. At this time there is no interface between these two networks.

Network support will basically be S-band with Ku-band utilized for special wideband data support. Full capabilities for S-band and Ku-band will exist through TDRSS subnet and several STDN sites (in addition to their full S-band capability) will be configured for Ku-band. The NASA Tug Onboard Communication Systems will be compatible with both the STDN and TDRSS.

Each STDN site will have full S-band capability with several prime sites configured for Ku-band. Each TDRSS satellite will have dual feed S-band/Ku-band 3.8 meter parabolic antennas plus a multi-element S-band array antenna, used primarily to relay communications to the Tug and payloads. A 1.8 meter parabolic Ku-band antenna will be used primarily for ground terminal transmissions.

Communications - The communications system consists of an airborne electronically steerable S-band phased array (AESPA) system, secure command decoder, and general purpose television camera. For rendezvous and docking a laser radar is used.

The AESPA system is a totally integrated S-band communications and tracking system that combines the functions of transmitting and receiving. The system performs the same function as a transponder and its associated antenna systems, but in addition can provide ranging and position location information. The AESPA can point and track a signal within a 120° cone normal to the array surface. The AESPA can handle both PCM telemetry and video data transmission simultaneously in a transmission mode and command and telemetry data in the receive mode.

The television camera system is a general purpose television system that employs both remote controlled pan and zoom of the camera system for observation. Future developments in technology are expected to provide a system that performs in both low level and normal lighting conditions.

A laser radar acquisition range of 50 n.mi. is assumed for SC retrieval operations.

Electrical Power System (EPS) - The Tug electrical power is furnished from two fuel cells, each rated at 2.0 KW with a 3.5 KW peak. Each fuel cell will be capable of supplying the total load. An auxiliary battery rated at approximately 25 amp-hr is provided to supplement for inrush current required for motor loads and to provide a means of powering up the fuel cells. It will also provide a fail-operate, fail-safe system. The estimated average power requirements for the Tug are 967 watts. An additional 90 watts are required for radar and TV during rendezvous and docking. In addition, 600 watts is allocated for supply to an attached SC. The steady-state voltage regulation will be 28 VDC (nominal). The Tug power system will not furnish voltages other than 28 VDC nor will it furnish any AC power. SC or equipment requiring AC power or voltages other than 28 VDC will need to supply these by means of built-in inverters/converters.

The fuel cells will not be started until Tug is released from the Orbiter. Power for the Tug and S/C will be provided by the Orbiter during the ascent phase. After release from the Orbiter the SC will receive power from the Tug.

Thermal Control System (TCS) - Passive and active TCS are employed to control Tug temperatures while the Tug is in any attitude.

Autonomy - The autonomy level of the various flight systems of the Tug are presently undefined. The following are the current

baselines for pertinent areas:

Shuttle Rendezvous - Automatically accomplished by onboard computation of G&N. All rendezvous to be coplanar with Shuttle, including abort. Beacon optional for Shuttle contact.

SC Rendezvous - Automatically accomplished by terminal phase guidance with cooperative target. Event telemetry (secure) monitoring.

SC Deployment and Monitor - Location, initialization, spin-up and release performed automatically. Prior to deployment, Tug monitors go/no go SC status. Thru puts SC telemetry to ground.

SC Docking - Accomplished automatically with target passive or not actively evasive. Event telemetry (secure) monitoring.

Mission Performance and Sequence - Two of the Tug geosynchronous missions will be (1) the delivery of multiple payloads to orbit, and (2) the delivery of a payload or payloads to orbit and retrieval of another payload or payloads. If in either of these missions, the payload longitudes should be different, then orbital phasing will be required to establish the correct longitude for each payload. The simplest phasing scheme consists of two equal, but opposite burns, separated by a coasting interval. If the desired longitude is behind the Tug, then the Tug would perform a burn to raise the semi-major axis of its orbit above geosynchronous altitude (geosynchronous altitude would be perigee of this orbit). This would create a differential orbital rate between the Tug and desired longitude which would cause the longitude to "catch up" to the Tug. The Tug would then circularize back to the geosynchronous orbit when it reached perigee of its phasing orbit. If the desired longitude is ahead of the Tug, then the Tug would lower its semi-major axis below geosynchronous altitude.

The delta-velocity required to phase is a function of the number of revolutions in the phasing orbit and whether the phasing is positive (Tug behind (west) of desired longitude) or negative (Tug ahead (east) of desired longitude). Figure A-6 shows delta velocity as a function of positive phase angle for 1, 2, and 3 revolutions in the phasing orbit. Figure A-7 shows the same data for the negative phase angle. Figure A-8

C-5

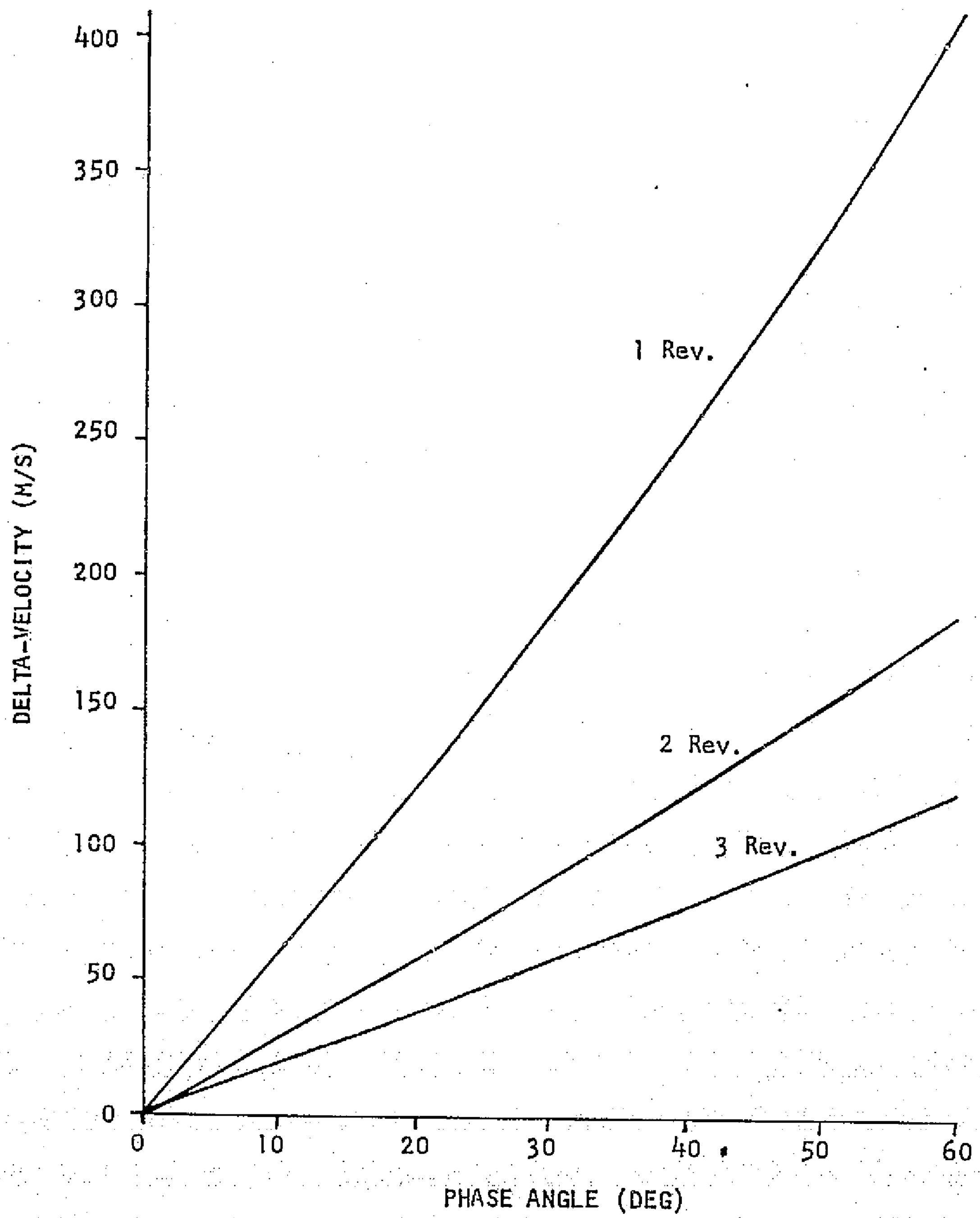

Figure A-6 Delta Velocity vs Phase Angle (Phasing from Below)

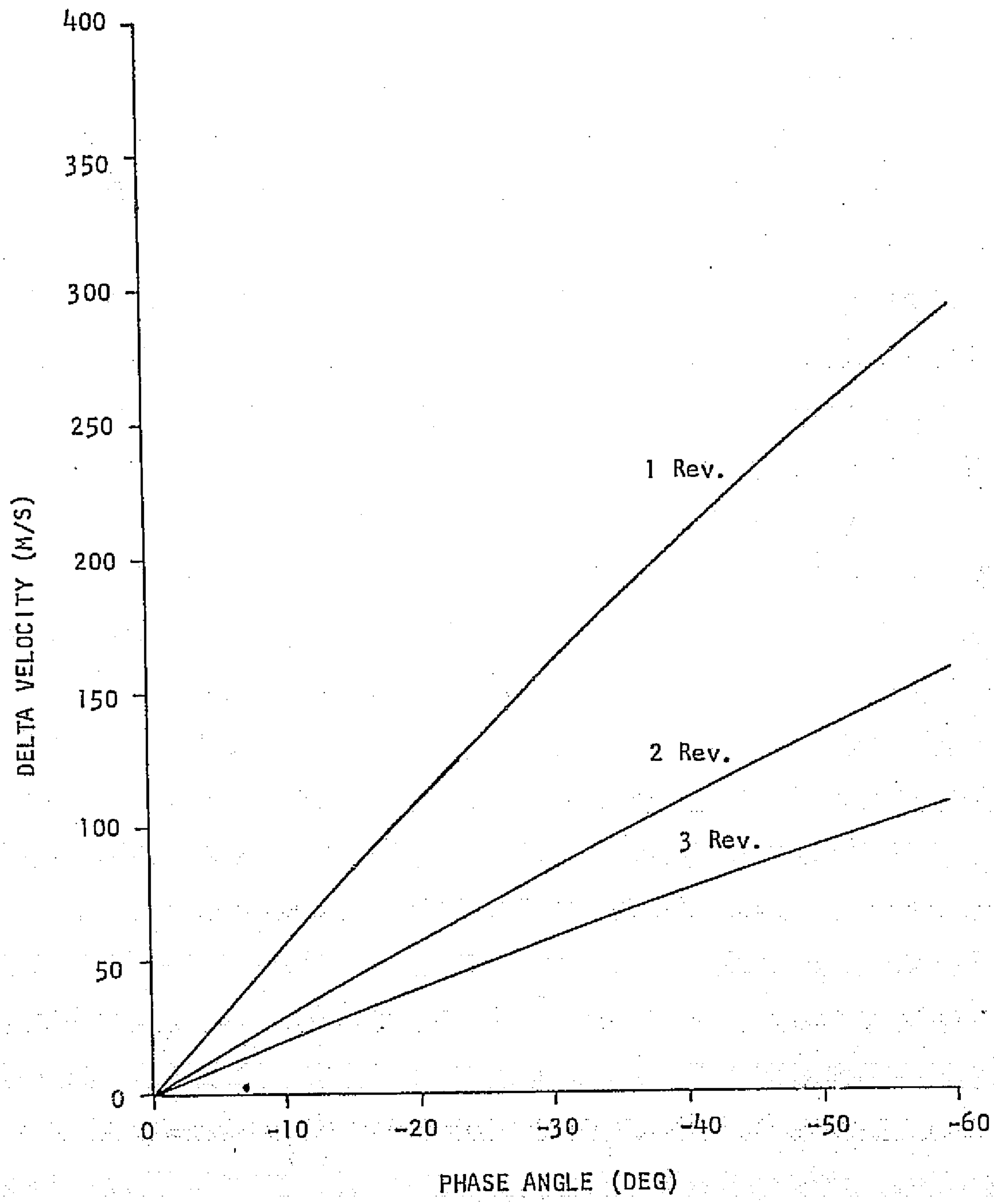

Figure A-7 Delta Velocity vs Phase Angle (Phasing from Above)

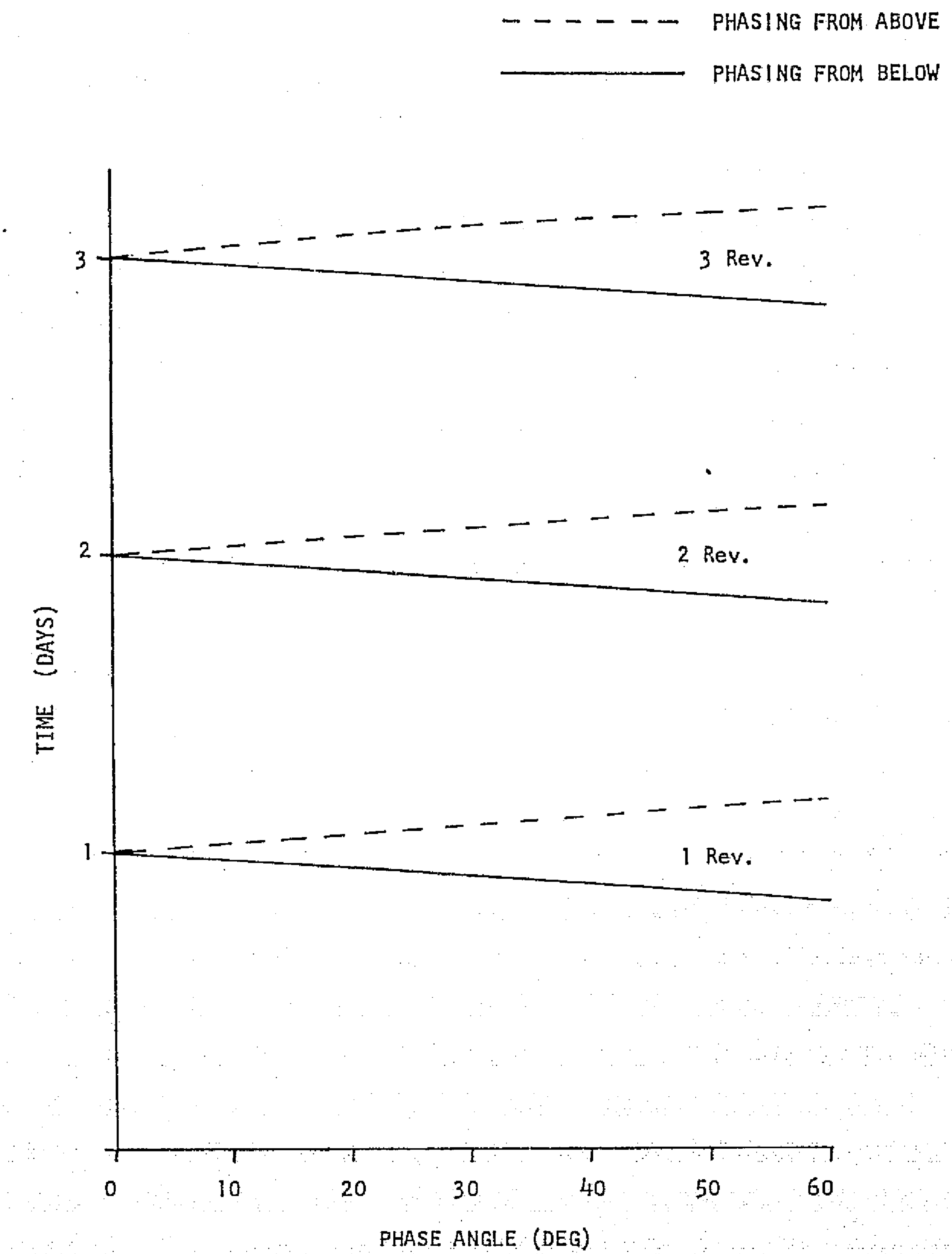

Figure A-8 Time Required for Phasing vs Phasing Angle

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	EVENT	INITIAL WT (lbs)	APS (lbs)	INERTS LOSSES	EVENT DURATION (Hrs)	TOTAL TIME (Hrs)	ΔV (F/S)
1	Tug Separation from Orbiter	58,855	8.6	10.0	2.0	2.0	
2	Phase in Shuttle Orbit	58,836	21.4	46.0	11.0	13.0	
3	Burn into Phasing Orbit (Full thrust)	58,769	-	-	0.13	13.13	4494
4	Coast in Phasing Orbit, One revolution	43,271	17.5	5.0	3.0	16.13	
5	Inject into Geosynchronous Transfer (Tank head idle + full thrust)	43,248	-	-	.11	16.24	3672
6	Coast to Midcourse Correction	33,678	13.8	6.0	1.5	17.74	
7	Midcourse Correction (Tank head idle + pump idle)	33,658	-	-	.03	17.77	50
8	Coast to Geosynchronous	33,536	14.0	16.0	3.96	21.73	
9	Circularize at Geosynchronous (Tank head idle + full thrust)	33,506			.12	21.85	5826
10	Coast and Orbit Trim	22,529	102.7	56	12.0	33.85	
11	Deploy First SC (1,000 lbs)	22,370	31.5	-	1.0	34.85	
12	Inject into Phasing Orbit (Tank head idle + pump idle)	21,338	-	-	.05	34.90	480
13	Coast in Phasing Orbit	20,623	13.7	105	28.0	62.90	
14	Circularize at Geosynchronous (Tank head idle + pump idle)	20,504	-	-	.05	62.95	480
15	Deploy Second SC (1,000 lbs)	19,816	37.7	6	1.0	63.95	
16	Inject into Phasing Orbit (Tank head idle + pump idle)	18,772	-	-	.04	63.99	258
17	Coast 1.5 Revolutions in Phasing Orbit	18,426	14.7	146	39.0	102.99	
18	Height Adjustment Burn (Tank head idle)	18,266	-	-	.01	103.00	10
19	Coast in Adjusted Phasing Orbit	18,251	11.1	49	13.0	116.00	
20	Phasing Orbit Circularization (Tank head idle + pump idle)	18,191	-	-	.04	116.04	258
21	SC Rendezvous and Retrieval (1200 lbs)	17,856	96.5	15	4.0	120.04	
22	Phase at Geosynchronous for Nodal Crossing	18,944	11.2	45	12.0	132.04	
23	Deboost Burn (Tank head idle + full thrust)	18,888	-	-	.08	132.12	5840
24	Coast to Midcourse Correction	12,686	7.5	6	1.0	133.12	
25	Midcourse Correction (Tank head idle)	12,673	-	-	.01	133.13	13
26	Coast to 170 n.mi. Perigee	12,659	8.1	24	4.2	137.33	
27	Inject into Return Phasing (Tank head idle + full thrust)	12,627			.05	137.38	3791
28	Coast 1 Revolution in Phasing	9,751	7.8	18	3.0	140.38	
29	Circularize at 170 n.mi. (Tank head idle + full thrust)	9,725			.05	140.43	4243
30	Rendezvous with Shuttle (SC - 1200 lbs. propellant reserve - 276 lbs)	7,280	32.4		4.0	144.43	

Figure A-9 Typical Mission Sequence for a Dual Deployment, Single Retrieval Geosynchronous Mission

shows the time required as a function of phase angle for both the positive and negative phase angle cases. If the phase angle is $+40^{\circ}$, then from Figure A-8, the delta-velocity required is 256, 120, 79 M/S for 1, 2, and 3 revolutions, and from Figure A-8, the time required would be 0.9, 1.9, and 2.9 days. If the phase angle is -40°, then from Figures A-7 and A-8, the delta-velocity required is 204, 108, 73 M/S, and the time required is 1.1, 2.1, and 3.1 days for 1, 2, and 3 revolutions in the phasing orbit. Figure A-9 presents a typical sequence for a dual deployment, single retrieval geosynchronous mission.

4. Earth Orbital Teleoperator Systems

The EOTS would be used for in-space assembly, maintenance, and repair of satellites. The EOTS is presently not baselined but laboratory models have been developed and there is an on-going concept development program. The EOTS will be approximately 4 ft diameter by 6 ft long in size. Operational range from the Orbiter will be 1 to 2 miles. The propulsion translation ΔV capability is 2000 ft/sec. The EOTS would use two TV cameras and manipulators with a 6 to 8-ft reach. (See Figure A-10).

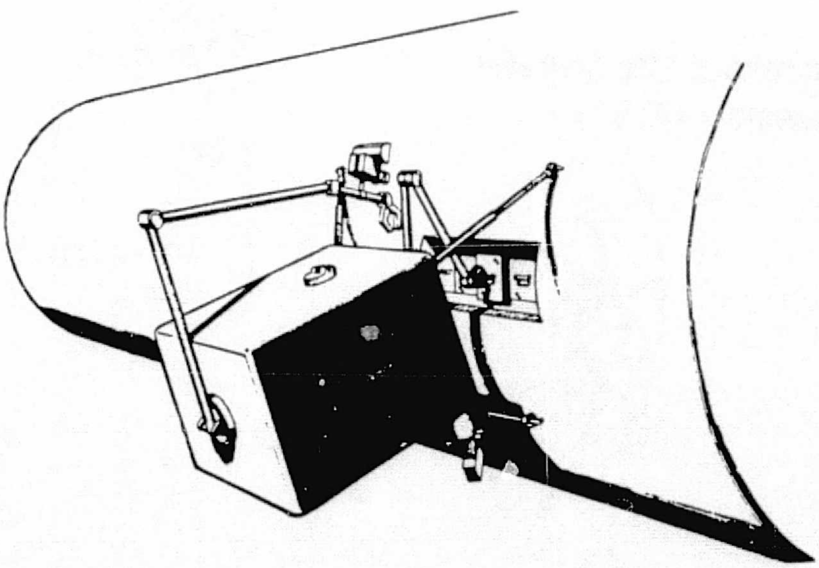

Figure A-10 Earth Orbital Teleoperator System

5. Manned Maneuvering Unit (MMU)

The MMU provides a self-contained propulsion capability to translate an extravehicular astronaut to and from worksites. The concept was proved in the Skylab M509 experiments. The MMU will be about 30 x 40 x 50 in. in size and weigh about 150 lbs. Cold gas (3000 psi N_2) capabilities are 50 ft/sec ΔV, but could be 300 ft/sec with hydrazine propulsion. The thrust levels are 5 lbs maximum. The automatic attitude hold capability is 4 ft-lb torque. The maximum flight duration before resupply is 6 hours (based on battery charge). (See Figure A-11.)

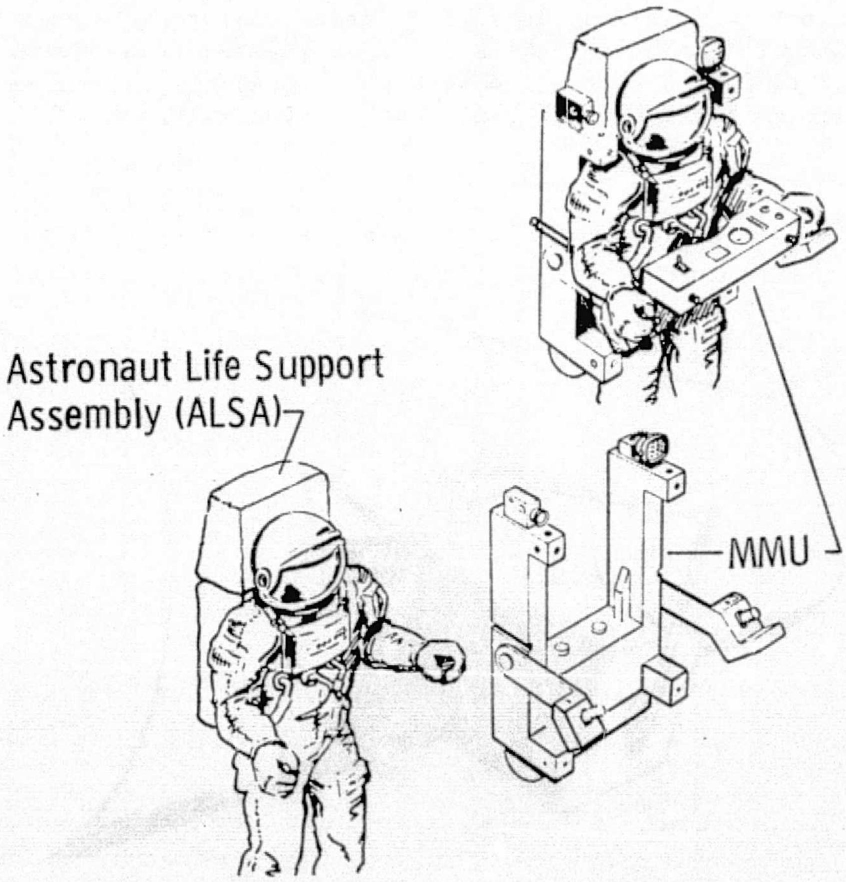

Figure A-11 Manned Maneuvering Unit

6. Shuttle Remote Manipulator System (RMS)

The RMS provides capabilities for payload handling and deployment, satellite capture, and satellite maintenance. The RMS is baselined on the Orbiter and is in concept development status. The RMS will have 6 degrees of freedom plus end effectors. During operation, both TV and operator's direct vision will be used. The RMS has a 50-ft total length and can provide 10 to 50-lbs tip forces. The maximum Shuttle payload (15 x 60 ft and 65,000 lbs) can be handled with the RMS. A second optional RMS would be chargeable to the payload. (See Figure A-12.)

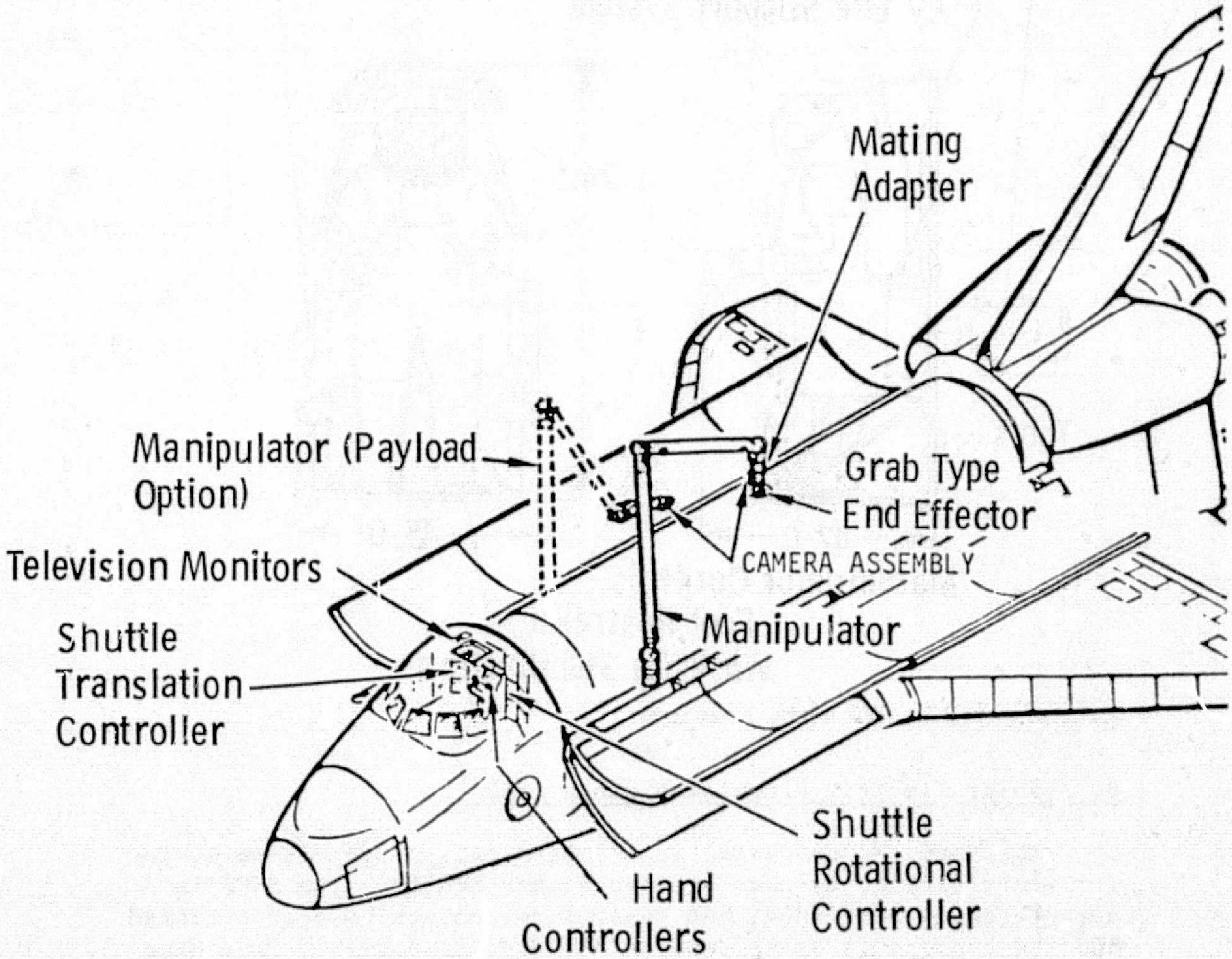

Figure A-12 Remote Manipulator System

7. RMS Work Platform

The RMS work platform would be used to position extravehicular crewmen at worksites. This item is presently only a concept and is not baselined. The platform would be about 76 x 45 x 42 in. in size. Accessories provided would be astronaut restraints, lights, manipulator controllers, and tool stowage. (See Figure A-13.)

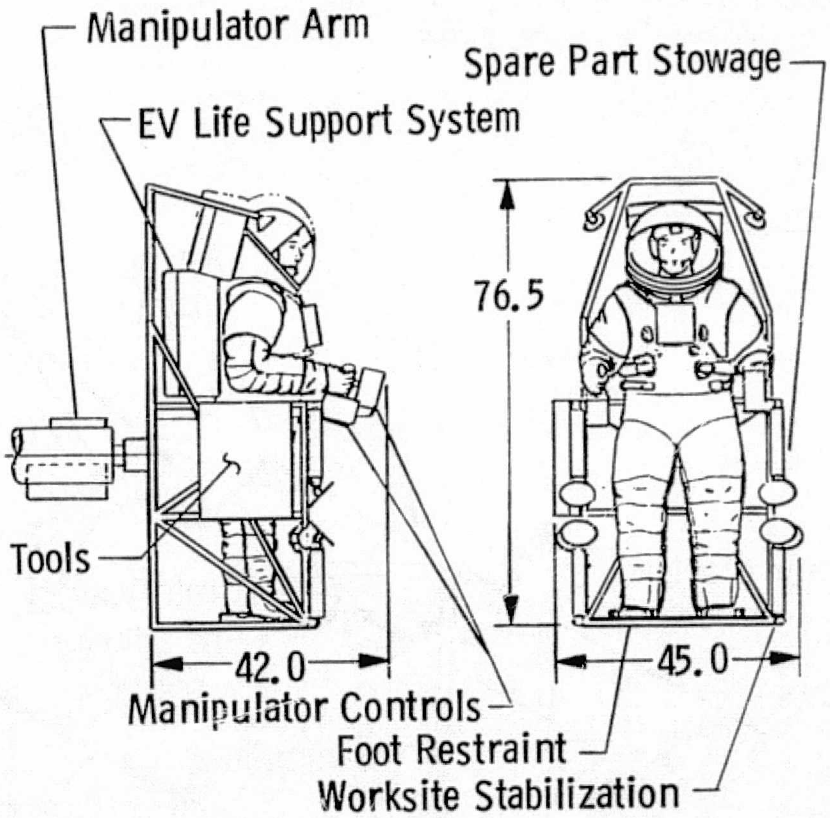

Figure A-13 RMS Work Platform

8. Solar Electric Propulsion Stage (SEPS)

The SEPS (Figure A-14) is a long-duration propulsive device for transferring payloads between higher orbits. The SEPS is not presently baselined, but several components have been tested and the concept is being studied. Thrust is obtained from nine mercury ion engines with characteristics of 0.206 lb thrust at 3000 sec I_{sp}. Electrical power for the engines is derived from solar arrays (25 KW initial capacity). With the arrays folded,

the SEPS is 10 x 15 ft in size. The dry weight is about 2800 lbs and mercury propellant weight is about 2900 lbs.

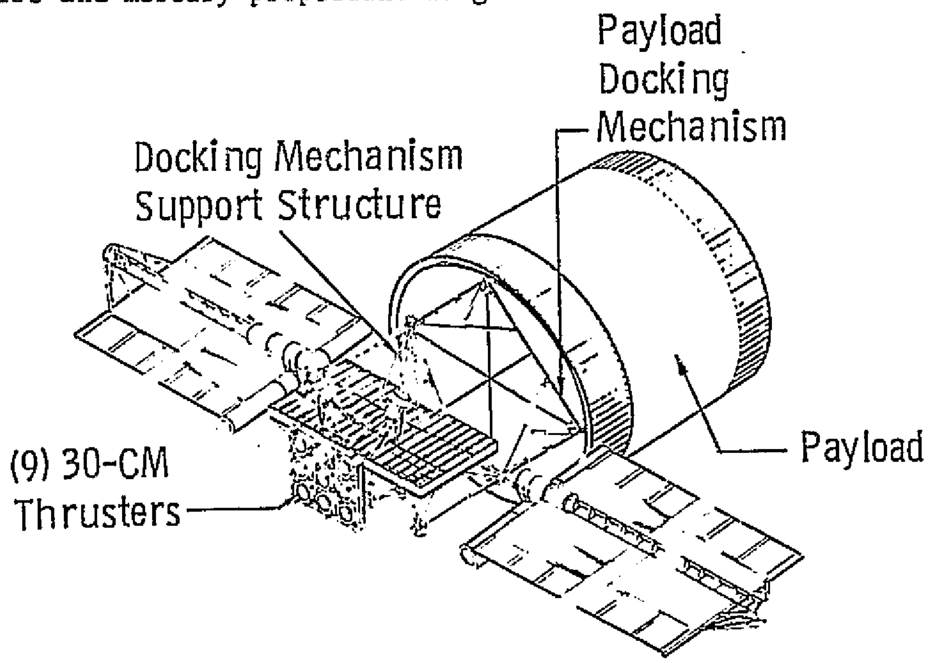

Figure A-14 Solar Electric Propulsion Stage With Typical Payload

9. Pressure Suit and Life Support

The status of pressure suits for the 1990 era is unknown. For purposes of this study, the Skylab A7L-B and ILC developmental suit data will be used. For these 4 psi suits, a 3-hour oxygen prebreathing time period is required. Of this time, 1.5 hours may be used for some of the EVA preparations. An additional 0.5 hr is required for final preparations. The allowable EVA duration is 6 to 8 hours. Post-EVA doffing and stowage time is estimated as 1.5 hours. (See Figure A-15.)

10. Manned Tug Module

The manned tug model (Figure A-16) is a preliminary concept that would permit manned maintenance, inspection, and assembly operations in higher orbits. The manned module would be about 8 ft long x 15 ft in diameter. The module would permit an orbital stay time of 5 to 7 days for four crewmen. Total weight

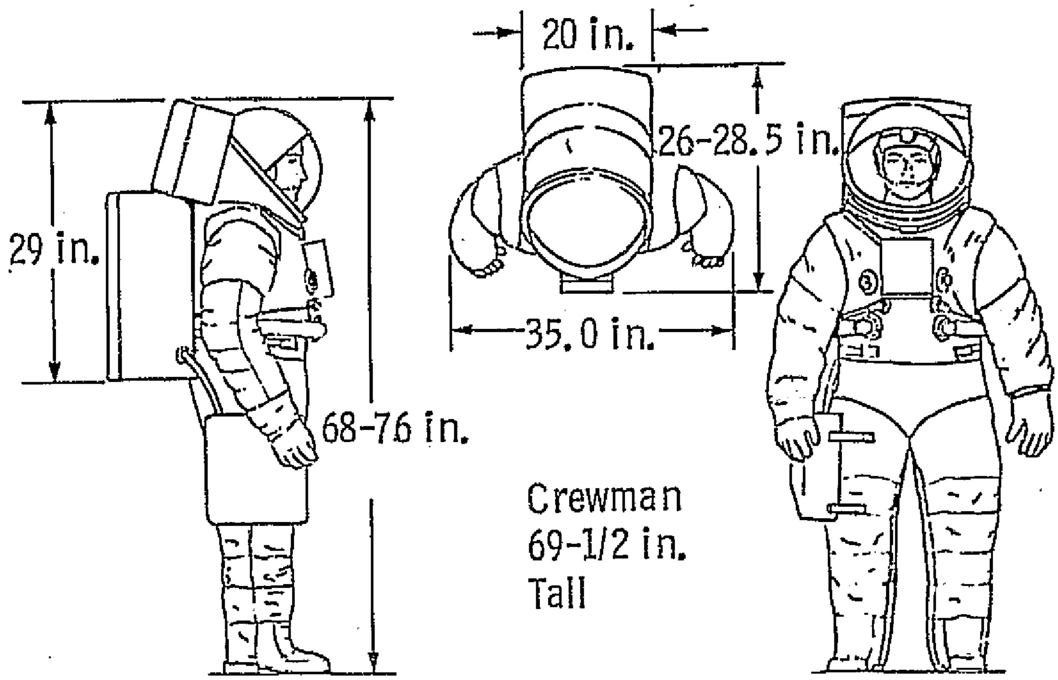

Figure A-15 Extravehicular Pressure Suit and Life Support

of the advanced Tug with manned module is estimated at 80,000 lbs (greater than Orbiter launch capability). Estimated weight (in lbs) of the manned module, from the Geosynchronous Platform Definition Study (Rockwell International), are:

	Stowage Tiers		
	1	2	3
Docking Mechanism	100	100	100
Primary Structure	800	900	1000
Secondary Structure	1450	1750	2050
Subsystems	3000	3000	3000
Tug Adapter	250	300	350
TOTAL (Lbs)	5600	6050	6500

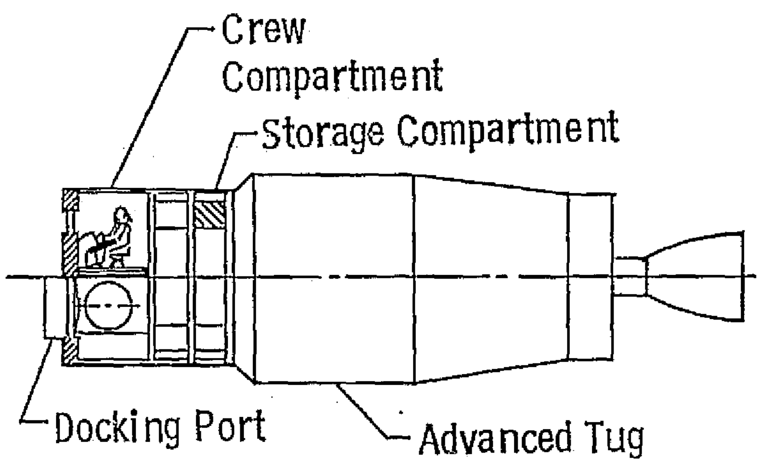

Figure A-16 Manned Tug Module

11. Data Reference Information

The following table gives the primary sources used to establish the above support equipment data.

Support Equipment	Primary Source
Shuttle Orbiter	Space Shuttle System Payload Accommodations, JSC 07700, Vol. XIV, Rev. C, July 1974.
Interim Upper Stage	Transtage IUS Overview, Martin Marietta Corporation, April 1974.
Tug	Baseline Space Tug documents, MSFC 68M00039, July 1974.
Earth Orbital Teleoperator Systems	Shuttle Remote Manned Systems Requirements Analysis, MCR-73-337, Martin Marietta Corp., December 1973.
Manned Maneuvering Unit	NASA Requirements Guidelines and Skylab M509 Performance.
Shuttle Remote Manipulator System	Space Shuttle System Payload Accommodations, JSC 07700, Vol. XIV, Rev. C, July 1974.
RMS Work Platform	Informal NASA material.

Support Equipment	Primary Source
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Pressure Suit and Life Support	Skylab data and ILC experimental suit data.
Manned Tug Module	Geosynchronous Platform Definition Study, SD73-SA-0036, Rockwell International, June 1973; and NASA discussions.

APPENDIX B

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3. Design/Cost Tradeoff Studies
4. Management Approach Recommendations
5. System Design and Specifications
6. Space Shuttle Interfaces/Utilization
7. Systems Definition Summary

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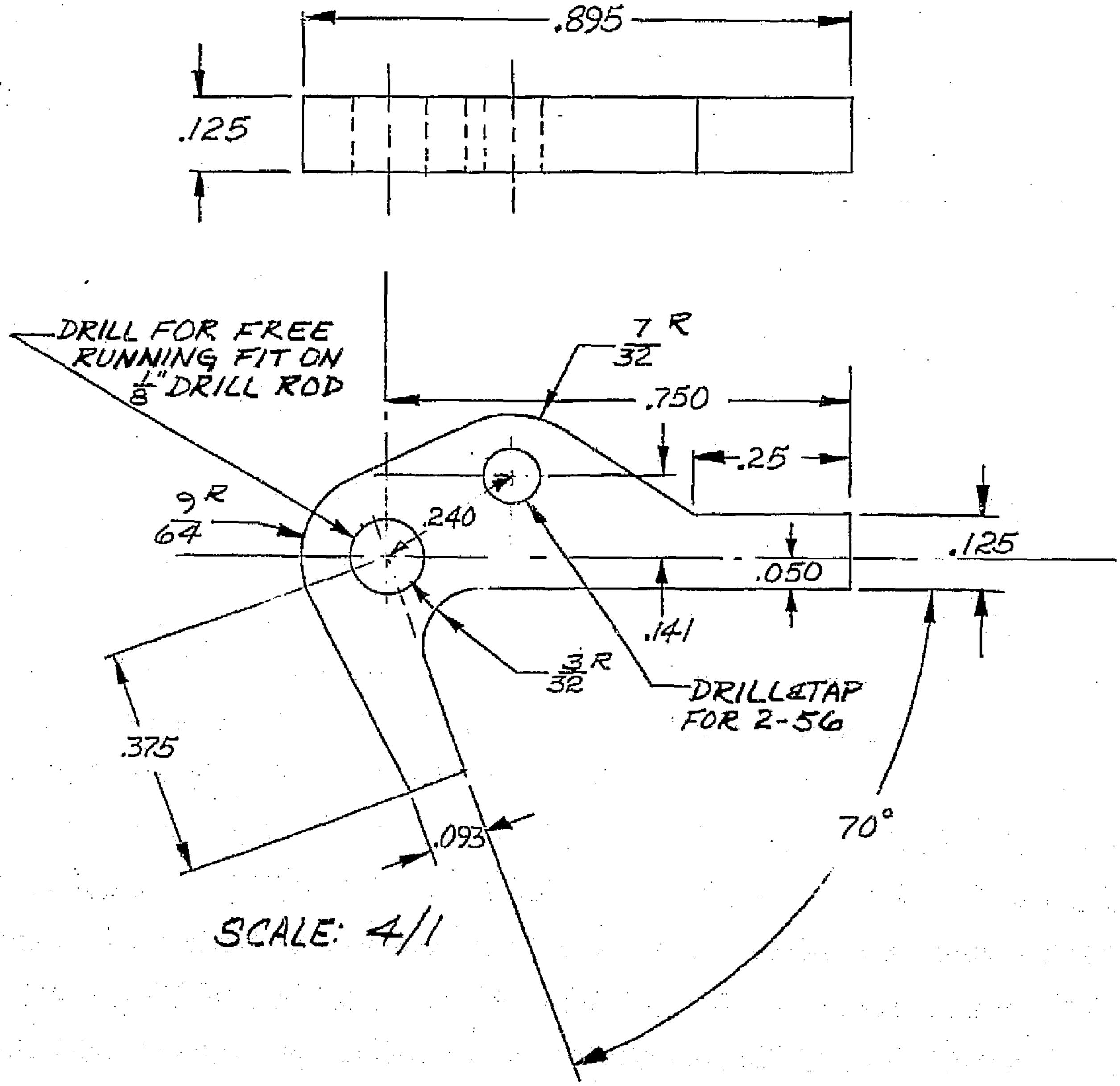
APPENDIX C

MECHANICAL ENGINEERING DRAWINGS

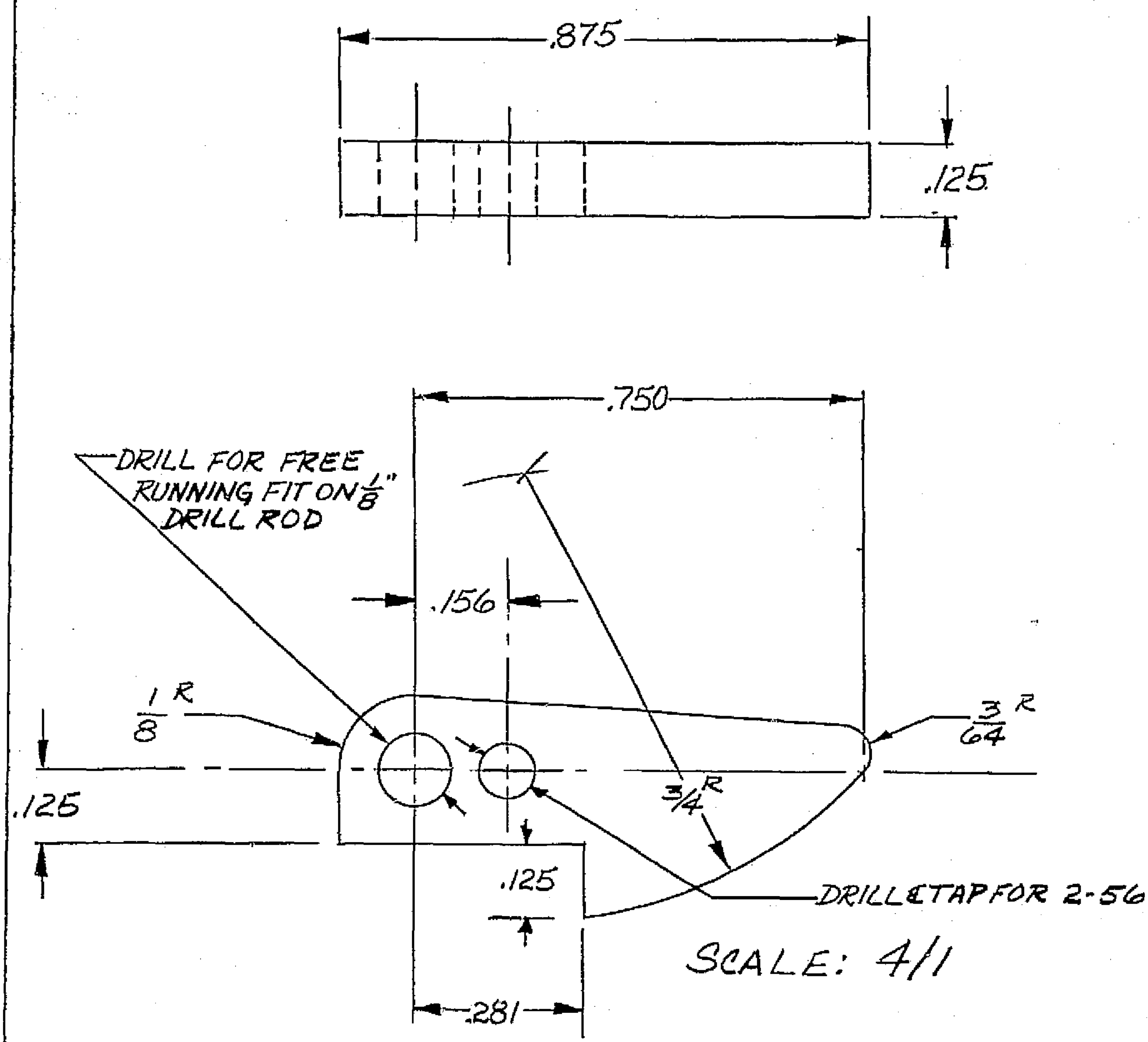
The mechanical engineering drawings for the latch mechanism for the RAT beam are shown here.



LATCH BLOCK, LOWER (2 REQ'D)

MAT'L: STEEL

LORP 2-11-75



LATCH BLOCK, UPPER (2 REQ'D)

MAT'L: STEEL

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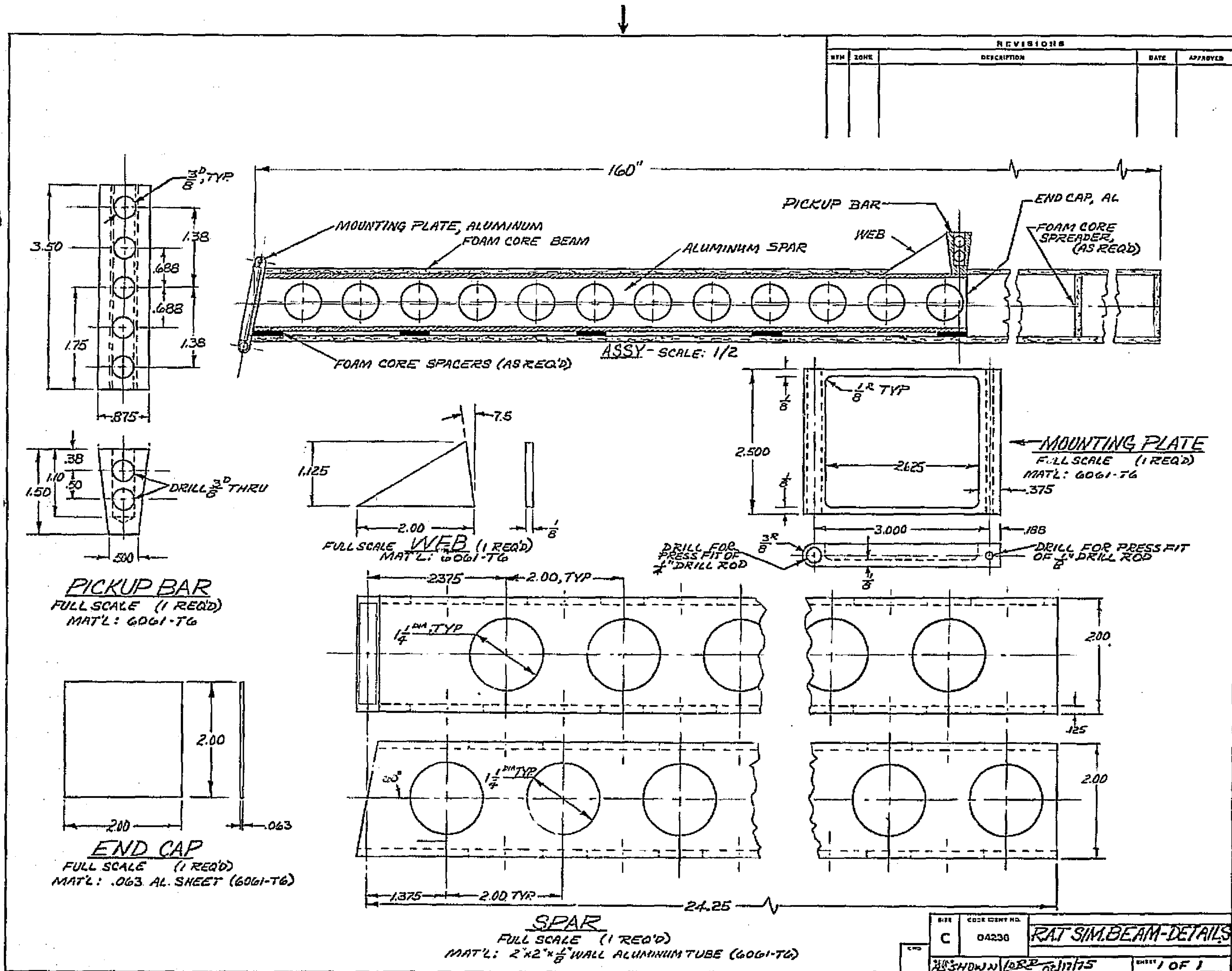
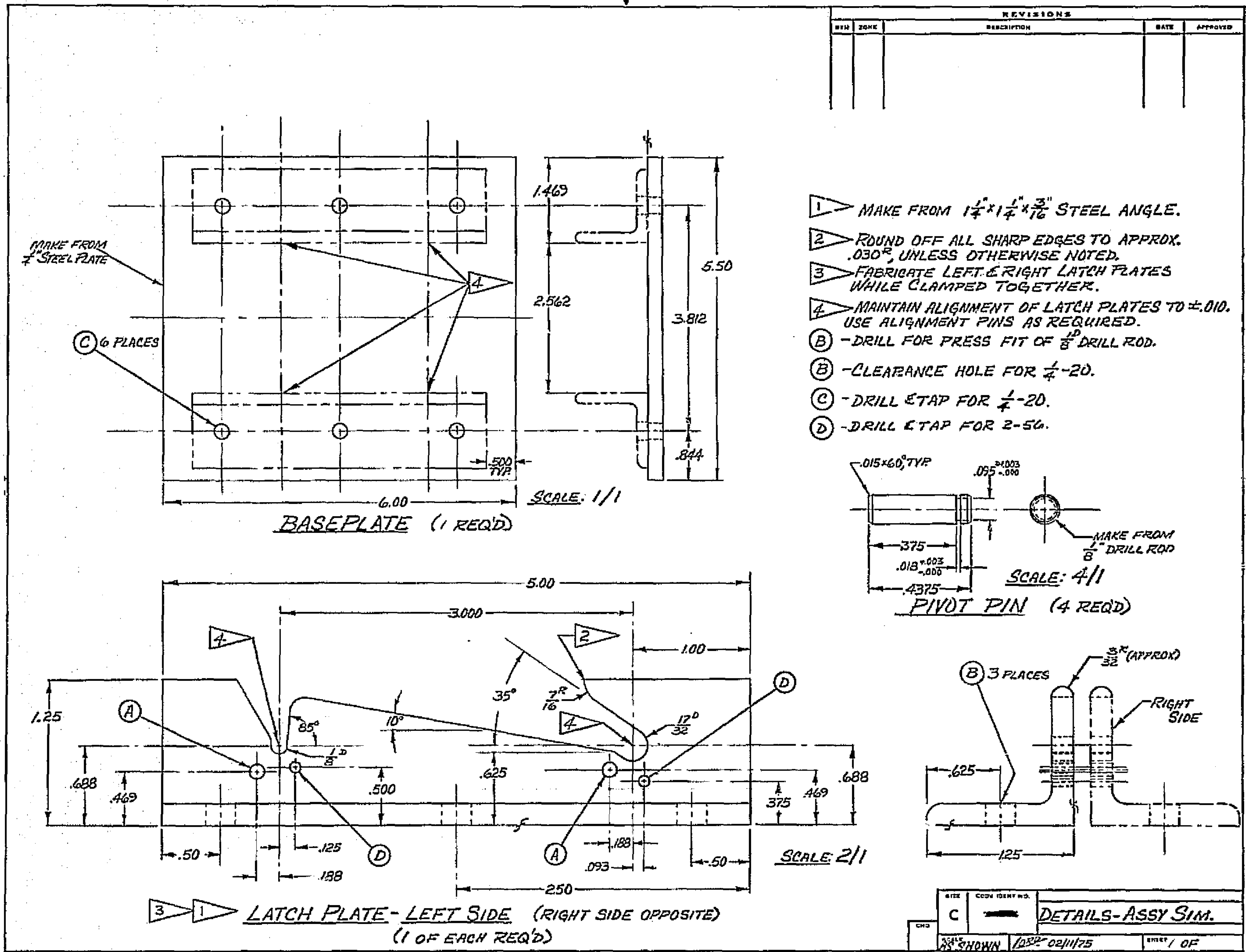
OF POOR QUALITY




REVISIONS
SYM | ZONE | DESCRIPTION | DATE | APPROVED
1 MAKE FROM $1\frac{1}{4}$" x $1\frac{1}{4}$" x $\frac{3}{16}$" STEEL ANGLE.
2 ROUND OFF ALL SHARP EDGES TO APPROX. .030R, UNLESS OTHERWISE NOTED.
3 FABRICATE LEFT & RIGHT LATCH PLATES WHILE CLAMPED TOGETHER.
4 MAINTAIN ALIGNMENT OF LATCH PLATES TO ±.010. USE ALIGNMENT PINS AS REQUIRED.
A - DRILL FOR PRESS FIT OF $\frac{1}{8}$D DRILL ROD.
B - CLEARANCE HOLE FOR $\frac{1}{4}$-20.
C - DRILL & TAP FOR $\frac{1}{4}$-20.
D - DRILL & TAP FOR 2-56.
MAKE FROM $\frac{1}{4}$" STEEL PLATE
C 6 PLACES
1.469
2.562
5.50
3.812
.844
.500 TYP
6.00
SCALE: 1/1
BASEPLATE (1 REQ'D)
.015x60° TYP
.095 +.003 -.000
.375
.018 +.003 -.000
.4375
MAKE FROM $\frac{1}{8}$" DRILL ROD
SCALE: 4/1
PIVOT PIN (4 REQ'D)
5.00
3.000
1.00
35°
$\frac{7}{16}$R
$\frac{17}{32}$D
85°
10°
$\frac{1}{8}$D
1.25
.688
.469
.500
.625
.375
.469
.688
.50
.125
.188
.188
.093
.50
2.50
SCALE: 2/1
B 3 PLACES
$\frac{3}{32}$R (APPROX)
RIGHT SIDE
.625
1.25
3 1 LATCH PLATE - LEFT SIDE (RIGHT SIDE OPPOSITE)
(1 OF EACH REQ'D)
SIZE C
CODE IDENT NO. [redacted]
DETAILS - ASSY SIM.
SCALE AS SHOWN
02/11/75
SHEET 1 OF